APOLLO

TECHNICAL MEMORANDUM NO. 15

RELIABILITY REVIEW OF

APOLLO GROUND TEST PROGRAMS (U)

Martin Co., Baltimore, Md.

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(Sponsored by NASA)

[6]

W. Ochsner,
Program Administrator
Apollo Reliability

W. Hadley, and
Manager
Apollo Reliability

R. Schwab
Asst. Tech. Dir.
Apollo Test

[1961] only

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APPENDIX - APOLLO RELIABILITY PROGRAM TASKS

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INTRODUCTION

The Apollo reliability effort emphasizes the successful accomplishment of seven technical tasks during the course of the complete program through lunar mission accomplishment. The scheduling control and integration of these seven tasks with other program efforts is essential to their successful accomplishment. This technical memorandum reports preliminary results for one of these seven tasks; a review of the Apollo ground test program to assess its ability to provide significant reliability data in the manufacturing phase of the program.

A description of all seven tasks of the Apollo reliability effort is contained in the appendix of the technical memorandum.

[redacted]

I. Test Designations

To provide for identification of all tests planned under the Apollo program, five designations have been established according to the basic purpose of the test. Test planning and test review functions are intended to assure the adequacy of tests under each of these five designations. A brief description of each designation is contained below:

1. Development tests:

   These tests are conducted on samples, models and prototype hardware to develop within the hardware, the capability to fulfill mission performance requirements. Hardware used for developmental tests is not suitable for use in flights.

2. Factory Qualification tests:

   These tests are conducted on flight configuration hardware manufactured to released drawings to demonstrate compliance with equipment specification requirements under anticipated flight environments. Included as a part of the qualification tests are any tests planned to demonstrate numerical requirements for equipment reliability. Hardware used for qualification tests is not suitable for use in flights.

3. Factory Acceptance tests:

   These tests are conducted on each flight article to assure suitability for flight use. These tests are also intended to show continuing compliance of each flight configuration article to the most important of equipment specification requirements.

[redacted]

4. Field Ground tests:

These tests are conducted on flight articles to prepare the complete system for flight test following receipt of factory-acceptance tested hardware at the field site.

5. Flight tests:

These tests are conducted on flight vehicles of configuration compatible with flight test objectives.

[redacted]

II QUALIFICATION TEST PROGRAM REVIEW

Qualification tests are tests run on prototype equipment to demonstrate that the engineering design and manufacturing processes are adequate to allow this equipment to meet its specification requirements. Qualification tests may include such tests as functional tests, structural tests, tests to failure, reliability tests, simulated environmental tests and combinations thereof. These tests can be performed on system, subsystem, component and/or part level.

The Apollo test program is being reviewed to assure that the following test considerations receive proper attention:

1. Sample size
2. Test Duration
3. Definition of failure or success
4. Criteria for acceptance
5. Data collection, processing, and analysis methods for tests yielding reliability data
6. Statistical confidence in conclusions drawn from the test results regarding equipment acceptability
7. Criteria for corrective action based on test results
8. Characteristics and measurement accuracy required of test facilities to obtain valid test results
9. Test conditions, including environment, for the equipment being tested
10. Provisions for demonstrating repeatability of test measurements

a) <u>Parts Qualification</u>

The current Apollo study phase is not expected to result in a detailed qualification program for parts. However, the ground rules and requirements will be specified and discussed in detail. Since no equipment can be better than the parts from which it is constructed, all Apollo parts will require qualification to standards commensurate with Apollo mission objectives and environment. Tests to failure, establishment of safety margins, studies of environments to which parts will be subjected, the effects of environmental stresses on parts and modes of failure analyses will be emphasized during development to assure that selection and application of parts receive necessary attention.

b) Components Qualification

Component qualification will be carried out at the component level and as discussed below, may be extended to testing at the subsystem level under actual rather than simulated system environments. Demonstration of reliability to reasonable statistical confidence levels, not feasible at the part level, will be considered in selected cases. In other respects component qualification testing will be conducted on a basis similar to parts qualification testing except that, in some cases, component qualification tests will be conducted in conjunction with subsystem qualification test.

c) Subsystem Qualification

Subsystem qualification testing affords the best means for demonstrating functional adequacy of the equipment with interaction of operating components under simulated environmental conditions. At this level demonstration of reliability to valid statistical confidence or risk levels can be carried out supplemented by performance variation tests of selected critical design parameters.

d) System Qualification

System qualification test will be conducted primarily under room ambient conditions; hot firings and tests under simulated critical environments will necessarily be conducted at lower levels. The primary objective of system qualification test will be to demonstrate functioning of the entire system within specified limits, both with respect to capability and repeatability. For both qualification testing and subsystem and component development and acceptance testing extensive use will be made of two special systems designed and built especially for test purposes. These are:

1. Laboratory Functional Mock-up
2. Complete System -Boiler Plate Structure

The functional structure will be a collection of easily accessible subsystems and components interconnected in as nearly identical fashion as the flight system. Its purpose will be to demonstrate operation of the system as a whole and the functional characteristics of its components and subsystems.

The boiler plate mock-up will duplicate as nearly as possible the flight system configuration. Selected components and subsystems may be brought outside the boiler-plate shell and subjected to critical environments during operation of the system. In every respect possible this system will be maintained to the latest engineering flight design configuration in order that test data will reflect the current design. It is expected that both these test tools will provide valuable data throughout the program, extremely useful in evaluating design improvements and in providing system reliability measurements.

e) Environmental Testing

The Figure "Spacecraft Subsystem Environmental Conditions for

Reliability Testing" illustrates at the subsystem level the type of environmental testing currently planned to demonstrate capability and reliability of the Apollo system. The various mission phases and subsystem operating and non-operating timer during these phases have been consolidated into six mission phases and further consolidated for test purposes where possible for each of eleven designated subsystems.

Qualification test considerations reflected in Figure block diagram include critical test environments, basic test duration, test sequence, environmental and subsystem test combinations. Sample size, actual test durations to demonstrate reliability and other required test considerations have not been finalized as yet. The diagram is intended to illustrate the approach which will be taken to develop a test program which will provide maximum dollar value and minimize test time.

Qualification Test Categories

The qualification test program will provide for four categories of tests to demonstrate capability and reliability assurance. These categories are:

1. "Standard" Environmental Testing
2. Performance Variation Tests (selected critical environments)
3. Safety Margin Testing
4. Statistical Demonstration Tests to specified confidence or risk levels.

1) "Standard" Environmental Testing

Categorized under "standard" environmental testing is the usual environmental tests under simulated operational environments - sand, dirt, moisture, vibration, acceleration, temperature, shock, etc. Certain of these environments, depending on the size of equipment under test and facilities available, will be combined, providing more realistic test results and reducing test time. These tests are designed to demonstrate capability of the equipment to meet its operational requirements.

The remaining three categories of tests are designed to demonstrate reliability of the equipment in various ways.

2) <u>Performance Variation Tests</u>

Performance variation tests will demonstrate the probability of equipment meeting selected critical design parameters. When high reliability requirements make numerical reliability demonstration testing extremely costly in time and sample size, performance variation tests, based on small sample sizes (for example, 10) have the advantage of providing, with acceptable statistical assurance, data upon which equipment performances can be predicted.

3) <u>Safety Margin Testing</u>

This type of testing provides valuable design information on equipment capability and provides a measure of the effect of critical environmental stresses on equipment life. It will be used extensively during development testing and will follow qualification testing under anticipated environments to provide additional assurance for critical applications.

4) <u>Statistical Demonstration Tests</u>

At higher system levels, where numerical reliability goals are compatible with available test timer, statistical demonstration to reasonable confidence or risk levels can be carried out under the move critical environments.

SPACECRAFT SUBSYSTEM ENVIRONMENTAL CONDITI[illegible]

SPACECRAFT SUBSYSTEM		1 COUNTDOWN 72 HRS.	2 BOOST & INJECTION 750 SEC.	3 CO[illegible] FLIGHT, [illegible] FLIGH[illegible] 332 HRS.	[illegible] POST, REENTRY MAN. 6 HRS.	REENTRY 1.2 HRS.	[redacted] 40 MIN.
1 STRUCTURES	OPER. TIME	72 HRS	12.5 MIN	338 HRS		1.2 HRS	40 MIN **
	CRITICAL TEST ENVIRON.	----	FLIGHT LOADS	LEAKAGE, INTERMITTEN VIBRATION TEMPERATURE (EXTERNAL & INTERNAL)		FLT. LOADS, TEMP. (LESS HEAT SHIELD)	LAND LOADS, PARASHUTE LOADS (DROP TESTS)
2 RECOVERY GEAR & COMMUNICATIONS	OPER TIME	1 HR *	12.5 MIN.	332 HRS.	6 HRS.	1.2 HRS.	----
	CRITICAL TEST ENVIRON.	ENVIRONMENTAL CONDITIONING IN NON-OPERATING MODE (LAUNCH PAD AMBIENT, FLIGHT LOADS, LEAKAGE, TEMPERATURE)					DROP TESTS (INCLUD. COMB. WITH STRUCT. DROP TESTS)
3 COMMUNICATIONS & TELEMETRY	OPER. TIME	2 HRS.	12.50 MIN.	173 HRS. OPER., 165 HRS. NON-OPER		1.2 HRS.	40 MIN
	CRITICAL TEST ENVIRON.	FLIGHT ENVIRONMENTS AFTER LAUNCH PAD CONDITIONING		LEAKAGE, INTERMITTEN VIBRATION TEMPERATURE (EXTERNAL & INTERNAL)		FLIGHT ENVIRONMENT, IMPACT LOADS	
4 GUIDANCE	OPER. TIME	2 HRS.	12.50 MIN	167 HRS.	6 HRS.	1.2 HRS.	
	CRITICAL TEST ENVIRON.	FLIGHT ENVIRONMENTS AFTER LAUNCH PAD CONDITIONING		LEAKAGE, INTERMITTEN VIBRATION TEMPERATURE (EXTERNAL & INTERNAL)		FLIGHT ENVIRONMENT IMPACT LOADS	
5 FLIGHT CONTROL, AUTO PILOT AND HOT GAS SYSTEM	OPER. TIME	0.1 HR.	12.50 MIN			1.2 HRS.	
	CRITICAL TEST ENVIRON.	AUTOPILOT – SAME AS GUIDANCE HOT GAS – SAME AS ATTITUDE CONTROL SYS.		FLIGHT CONTROL, AUTO-PILOT, HOT GAS } 338 HRS			
6 ATTITUDE CONTROL AND VERNIER SYSTEM	OPER. TIME	0.1 HR. *	12.50 MIN. *	ATTITUDE CONTROL AND VERNIER SYSTEM } 338 HRS			
	CRITICAL TEST ENVIRON.	STATIC VIBRATION LAUNCH PAD ENVIRON. CONDITIONS. EQUIP. NON-OPER. HOT RUN BEFORE & AFTER TEST		TEMPERATURE, SELF GENERATED FLIGHT LOADS			
6a MISSION CONTROL PROPULSION	OPER. TIME	0.1 HR. *	12.50 MIN *	150 SEC., 338 HRS. *		1.2 HRS. *	40 MIN *
	CRITICAL TEST ENVIRON.	STATIC VIBRATION LAUNCH PAD ENVIRON. CONDITIONS. EQUIP. NON-OPERATING HOT RUN BEFORE & AFTER TEST		MULTIPLE STARTS AT AEDC (VACUUM, TEMP. STATIC FIRING USING COMMAND MODULE.			
7 POWER SYSTEM	OPER. TIME	3 HRS.	12.50 MIN.	338 HRS.		1.2 HRS.	40 MIN.
	CRITICAL TEST ENVIRON.	SOLAR CELLS – SAME AS STRUCTURE APU OF FUEL CELLS – SAME AS REACTION CONTR.		SOLAR CELLS: VACUUM, SOLAR HEAT APU OR FUEL CELLS: CONDITIONED UNDER CONDITIONS (3) & (4), OPERATED UNDER (5)			BATTERIES OR FUEL CELLS OPERATED AFTER CONDITIONING UNDER (1) → (5)
8 DISPLAYS, CONTROLS, CREW	OPER. TIME	3 HRS.	12.50 MIN.	332 HRS.	6 HRS.	1.2 HRS.	40 MIN.
	CRITICAL TEST ENVIRON.	FLIGHT SIMULATOR TESTS		FLIGHT SIMULATOR TESTS		FLT. SIM. TESTS	MANNED AIR DROPS
9 ENVIRONMENTAL CONTROLS	OPER. TIME	3 HRS.	12.50 MIN.	332 HRS.	6 HRS.	1.2 HRS.	
	CRITICAL TEST ENVIRON.	PAD & BOOST ENVIRONMENT WITHOUT STRUCTURE		LEAKAGE TEMPERATURE RADIATION (BOILER PLATE STRUCTURE		REENTRY ENVIRON. WITHOUT STRUCTURE	
10 LAUNCH ESCAPE SYSTEM	OPER. TIME	30 SEC					
	CRITICAL TEST ENVIRON.	INSTALL ON PAD AND PREFLIGHT CHECKOUT	DEMONSTRATE OFF PAD MAX. q & MAX. q				

* NOT OPERATING

** 71 HRS. POST LANDING OPERATION NOT INCLUDED IN TABULATION

[redacted]

III. Flight Article Acceptance Test Program

The ability of the factory acceptance test program to provide data for measuring numerical reliability will be one of the areas of intensive review during the detailed planning of acceptance tests. The normal pattern of factory acceptance testing follows a building-block concept in which parts, components, sub-systems, and ultimately, the complete vehicle are subjected to tests which assure the suitability of each manufactured item for use in flight. At this stage of the Apollo program sufficient detail does not yet exist in the factory acceptance test plan to assess with precision the exact amounts of data which will be available for measurement of reliability. The extensive experience of the Martin Company on related programs does, however, permit significant conclusions to be drawn regarding the measurement of reliability at each of the building-block steps of a factory acceptance test program. These conclusions are stated below:

(a) Parts Acceptance Testing

Factory acceptance testing at the parts level is not expected to yield any significant demonstration of numerical reliability required of parts for use in the spacecraft. The two factors which prevent demonstration at this level are the amount of data required for demonstration and the adequate simulation of part environments including application factors and interaction with other parts. Numerical reliability requirements of parts for use in the spacecraft are normally in the order of 1,000,000 hours MTBF (or 10^6 duty cycles MTBF). Data at this level are, however, essential to the control of parts adequacy and to the elimination of detected problems.

(b) Component Acceptance Testing

Factory acceptance testing at the component level falls much in the same category as parts in yielding a significant demonstration of numerical reliability requirements. Although much less data is required for component reliability demonstration because the numerical requirements average one or two orders of magnitude lower system application and interaction factors are still not present during tests. Components do, however, normally undergo simulated environment tests as a part of acceptance yielding valuable data on environmental adequacy and improvements required.

(c) Sub-system Acceptance testing

Factory acceptance testing at this level of equipment complexity normally provides an adequate index of numerical reliability when the results of several systems worth testing are combined. Subsystem interaction factors are usually missing, however, as well as valid simulation of flight environment. This type of testing is most valuable in detecting variances in numerical reliability between identical subsystems and in detecting sub-systems of significantly lower-than-required reliability. For the Apollo subsystems which operate continuously during the translunar and transearth phases of flight, the amount of testing for acceptance will not provide numerical reliability indices of significance.

(d) Spacecraft Acceptance Testing

Factory acceptance testing of composite subsystems installed in the spacecraft will provide an index of numerical reliability for the group of subsystems operated. Under normal spacecraft acceptance testing, however, the propulsion subsystems would not be operated except for their electrical and pneumatic sequencing and their effect on the reliability of the remaining subsystems would therefore not be determined. Simulation of flight environments is also not practical at this level thus indices would be determined under factory ambient conditions. The major benefits would be in assessing full subsystem interaction effects on reliability of the systems tested and in determining crew reliability when subsystems are sequencing through their normal mission functions. It would be expected that the simulation of a spacecraft mission during acceptance would shorten the amount of time in the translunar and transearth phases of flight.

(e) Additional Sources of Data Supplementing Factory Acceptance Tests

Two primary additional sources of data will exist during the factory acceptance test program. These are the continuing tests of functional system mockups and a functional spacecraft. These equipments will continue in test, following their use for development purposes, paralleling the factory acceptance test program. The primary purpose of these equipments is in their use for failure evaluation and for testing of proposed changes. The nature of this testing is expected to prevent unrestricted use of data generated for evaluation of reliability, however, those deviations from flight configuration and from fixed test procedures will be carefully recorded to prevent erroneous conclusions concerning reliability. The major contributions of these equipments to reliability data acquisition are therefore expected to be in providing data on problems not anticipated during latter phases of the program.

IV Check List for Test Review

The review of test plans to assess their ability to yield significant reliability data is a most important function at this stage of the Apollo program. Substantial influence can be exerted to make each test yield a maximum amount of data without significant increase in cost. The basic approach used by the Martin Company on similar programs is to minimize added-cost reliability testing and to derive most of the data from normally scheduled testing. In accordance with this concept, a detailed review of normally scheduled testing is conducted with emphasis directed on the following test parameters:

1. <u>Sample size and test duration</u>

   These parameters are reviewed to assure data on variance between samples and to assure that test durations simulate mission times. The appropriate balance between sample size and test duration can also provide data on equipment life without resort to specially planned life tests.

2. <u>Criteria for acceptance</u>

   An essential part of each test program which is frequently overlooked in the test plan is the establishment of acceptance criteria, including a definition of success and failure. Test review insures that each planned test has adequate definitions in this area.

3. <u>Statistical confidence of conclusions</u>

   Review in this area is directed at assessing the accuracy of measurement considering both basic measurement system capability and the number of replications of critical measurements.

4. <u>Operating characteristic curves</u>

   Emphasis on this test parameter is directed at ascertaining the probability of erroneous conclusions from the test results.

5. <u>Test conditions</u>

Review of this test parameter is directed at assuring adequate simulation of mission environments and adequate adherence to the ordering of tests to simulate the mission sequence.

6. <u>Data acquisition and processing</u>

Review in this area insures collection of required reliability data and expeditions processing to meet program schedule requirements.

7. <u>Criteria for corrective action</u>

Review of the test plan in this area insures that a prescheduled sequence of events is followed in event of failure during test and that failure data is fully utilized to effect reliability improvements.

APPENDIX

A. RELIABILITY

The Apollo reliability effort is a technical task directed at elimination of trouble. During the study contract, problem areas and weaknesses have been anticipated and corrected by means of evaluation of proposed designs. During subsequent program plans, emphasis will change to the analysis of test data and the solution of known problems. This present concentration on evaluation of design, coupled with careful planning for subsequent program phases, is essential to the conduct of a successful reliability effort on Apollo.

The tasks which must be performed on subsequent phases have been identified and form an important part of the overall program plan.

During the present Apollo contract, reliability work has concentrated on:

(1) Support of the design effort to arrive at a configuration with high inherent reliability.

(2) Support of the program planning effort to define the conduct of a follow-on contract for the system.

As a result of this effort the technical and management aspects of the reliability program have received serious attention.

The program comprises seven tasks:

1. <u>Determination of Numerical Reliability Requirements</u>

Values of reliability consistent with the desired probability of mission accomplishment must be established as a basis for evaluating proposed systems.

2. <u>Definition of Design Requirements</u>

The required numerical values of reliability must be translated into requirements which the designer can fulfill through known design techniques. At the present state in reliability, only gross rules are available to identify design features which will yield given numerical values of reliability. Considerable negative data is available, however, to identify design features which will not yield a specific required value of reliability.

3. <u>Conduct of Design Evaluation Studies</u>

Alternate methods of performing a system function must be evaluated to select the superior method. Reliability effort in this area yields high dividends because the emphasis is on comparative analysis rather than on absolute values of predicted reliability.

[redacted]

4. Conduct of Design Review

For the selected system design, compliance or non-compliance with design requirements must be documented. Most significant is the identification of requirements for which compliance cannot be ascertained when the design review is in progress. The program should be carefully reviewed at this point to provide proof of compliance in these unknown areas prior to flight.

5. Identification of Critical Product Characteristics

As part of design review on the selected configuration, this task provides a means of controlling the product during procurement, manufacture, shipping, handling and storage. Failure to identify significant product characteristics will result in generalized types of control methods in the above areas which do not anticipate causes of product degradation.

6. Demonstration of Control of Critical-Product Characteristics

Every inspection or test performed on an article of flight configuration hardware is intended to prove that certain product characteristics are under control. This task insures that inspection and testing does, in fact, provide such proof.

7. Reliability Data Collection and Utilization

Reliability data on tests conducted, equipment operating time and equipment discrepancies must be collected and analyzed to determine status in achieving system reliability and to provide an organized method for improving system reliability in the areas yielding the biggest dividends.

The management aspects of the program are associated with accomplishing these seven tasks on schedule and insuring that the information resulting from each task is used to influence decisions in the conduct of the program.

How these tasks apply to Apollo may be seen in Fig. IV-7, which shows the key events during the life cycle of the Apollo vehicle, the program activity required to make each event occur and the specific reliability effort which takes place as a part of the program activity.

The present reliability effort on Apollo will result in a specification which documents requirements for the conduct of the Apollo reliability effort, a program plan showing the specific tasks which will meet the requirements of the specification and the detailed reliability analysis performed on the spacecraft systems. Subsequent figures and text show the nature and format of the system analyses which have been conducted since program inception.

[redacted]

PROGRAM EVENT	PROGRAM EFFORT	RELIABILITY EFFORT
CONTRACT GO-AHEAD		
	ESTABLISHMENT OF MISSION OBJECTIVES. FORMULATION OF DESIGN REQUIREMENTS	ESTABLISH NUMERICAL REQUIREMENT FOR MISSION ACCOMPLISHM DEFINE MISSION SUCCESS AND CREW SAFETY ALLOCATE REQUIREMENTS TO MAJOR SYSTEMS TRANSLATE NUMERICAL REQUIREMENTS TO REQUIRED DESIGN FEATURES
ISSUE VEHICLE DESIGN REQUIREMENTS		
	EVALUATION OF ALTERNATE DESIGNS SELECTION OF FLT. VEHICLE CONFIGURATION	EVALUATE FAILURE RATES OF ALTERNATE DESIGNS DEFINE POTENTIAL RELIABILITY TROUBLE AREAS SUGGEST IMPROVEMENTS OR ALTERNATE CONFIGURATIONS IDENTIFY CRITICAL CHARACTERISTICS OF SELECTED DESIGN DOCUMENT CONTROL METHODS FOR CRITICAL CHARACTERISTICS
RELEASE MANUFACTURING DRAWINGS FOR SELECTED CONFIGURATION		
	PROCUREMENT AND MANUFACTURE OF HARDWARE	CERTIFY CONTROL OF CRITICAL CHARACTERISTICS DETECT AND CORRECT UNANTICIPATED CHARACTERISTICS
DELIVER HARDWARE FOR QUALIFICATION TEST		
	QUALIFICATION TESTS OF FLIGHT CONFIGURATION HARDWARE	SPECIFY TEST REQUIREMENTS FOR QUALIFICATION ANALYZE TEST RESULTS FOR COMPLIANCE WITH REQUIREMENTS DETECT AND CORRECT UNANTICIPATED PROBLEMS EVALUATE ADEQUACY OF IMPROVEMENTS
CERTIFY VEHICLE HARDWARE DESIGN COMPLIANCE		
	ACCEPTANCE TEST AND SHIPMENT OF FLIGHT VEHICLE HARDWARE	SPECIFY ACCEPTANCE TEST REQUIREMENTS ANALYZE TEST RESULTS FOR COMPLIANCE WITH REQUIREMENTS DETECT AND CORRECT UNANTICIPATED PROBLEMS EVALUATE ADEQUACY OF IMPROVEMENTS
FLIGHT VEHICLE SYSTEMS: DELIVERY TO TEST SITE		
	PREFLIGHT CHECKOUT OF FLIGHT VEHICLE SYSTEMS	CERTIFY CONTROL OF CRITICAL CHARACTERISTICS DETECT AND CORRECT UNANTICIPATED PROBLEMS EVALUATE ADEQUACY OF IMPROVEMENTS
FLIGHT VEHICLE DELIVERY TO LAUNCH PAD		
	VEHICLE INSTALLATION, CHECKOUT AND COUNTDOWN FOR FLIGHT	DETECT AND CORRECT UNANTICIPATED PROBLEMS EVALUATE ADEQUACY OF IMPROVEMENTS
VEHICLE LAUNCH		
	LUNAR MISSION	ANALYZE RESULTS FOR COMPLIANCE WITH MISSION OBJECTIVES
CREW RECOVERY		

[redacted]

[redacted]

RELIABILITY EF

Flight Test & Crew Training

TM-18

Test Logic Analysis

For each plan considered, a test logic analysis has been made in order to fully investigate all expected problem areas that could be encountered and to formulate an approach to the solution. As a basis for this analysis, two assumptions were made, one, an expected overall test result probability and a percentage of data return that would validate proceeding to the next test objective. It is well to note that these figures though based on assumptions are fairly realistic numbers. From these figures we were then able to determine the vehicle requirements that would satisfy our test objectives. Ref. Fig. 18-2a.

Figures 18-1a thru 18-1c are a general and detailed logic plan used in determining our operational philosophies during the test program. In this manner, we have retained the ability of making early decisions in problem areas without serious delays to the program.

Since we have not "per se" assigned specific vehicles as back-up throughout the program, we have aligned our testing process so that there is an established plan for each contingency. In this manner we can avoid the usual program delays and stoppages.

Obviously we cannot, in the limited space available, detail each test series and phase therefore we shall only outline all the procedures and philosophies used for a single test within a series and phase. The sample used in Fig. 18-1d is for PHASE III Series 1, Undershoot Trajectory. For this outline we will consider only three general result categories. (Actually the test results probabilities are more than three.)

Case 1. Success - The conclusions here are self explanitory.

Case 2. Booster Malfunction - This maybe the result of abort, range safety or improper flight profile etc. It could also be considered as incomplete data though any test will furnish some usable results.

Case 3. Spacecraft Malfunction - This is considered a discrepancy also in that the vehicle will have arrived in its area of test but test results are lacking because of limited data, system malfunction or no data at all.

In Case 2 then we with NASA, AMR and booster contractors make an analysis based on data obtained. For the most part the test objectives are rescheduled for the subsequent vehicle with its test objectives transferred to the next. It is well to note here that the sample used for outline is the most critical test area in that we cannot double up the mission objectives of the next test.

Case 3 also requires an analysis by the contractor, his associates and NASA based on data return. Based on the results the required changes are incorporated into the next test article and rescheduled. Where the changes are major in nature facilities may not be adequate to incorporate them into the next test vehicle at the launch site. Should this be the case these changes would be started in the earliest vehicle possible that would allow the minimum of test time slippage.

Having made the above decision we then reschedule the mission. As before we can expect, though improbable, the results that occurred in Case 2 and Case 3. Should however, this occur we then carry our analysis into its second step.

Step II

Case 2. The reasons for arriving here will remain in all probability the same, our course of action however will differ somewhat. The Test

Review Board together with NASA, Booster Contractors and AFMTC will now make a critical review of mission objectives in terms of booster, systems and range capability. Based on these results the earliest possible decision is made as to our next step which could be additional boosters and spacecraft if we are to test to success (80% data return probability) or test compromises that can be made without jeapardizing the test program.

Case 3. Here as in the first step of the analysis the reasons will for the most part remain the same. The Test Review Board together with NASA and sub-contractors will analyze test results, factoring it by a critical systems analysis, manufacturing procedures and supporting tests that are used in arriving at our previous changes. They will then make a decision as to system or systems redesign or rework, their testing and our approach to future testing. If testing to success is mandatory, based on test results we then again have at the earliest possible time made a decision as to additional booster and spacecraft allocation. Or as in Case 2, we can, based on limited data, arrive at a test compromise or mission objective tradeoff that will allow us to proceed.

There are other facets of this analysis too numerous to cover here but it is worth metioning some key highlights.

1. Each change will be backed by through and comprehensive testing.
2. When proceeding from one series or phase to the next series or phase the first vehicle in each is for the most part compatible to the previous one.

3. The Test Operations Staff by being "on the scene" insures a continuity of effort and test throughout the program. Since a secondary function of this staff is to act as the Test Review Board they have a complete and through knowledge of the problem areas.

APOLLO IMPLEMENTATION PLAN
GENERAL LOGIC ANALYSIS
PHASE I & II

Fig 18-1a

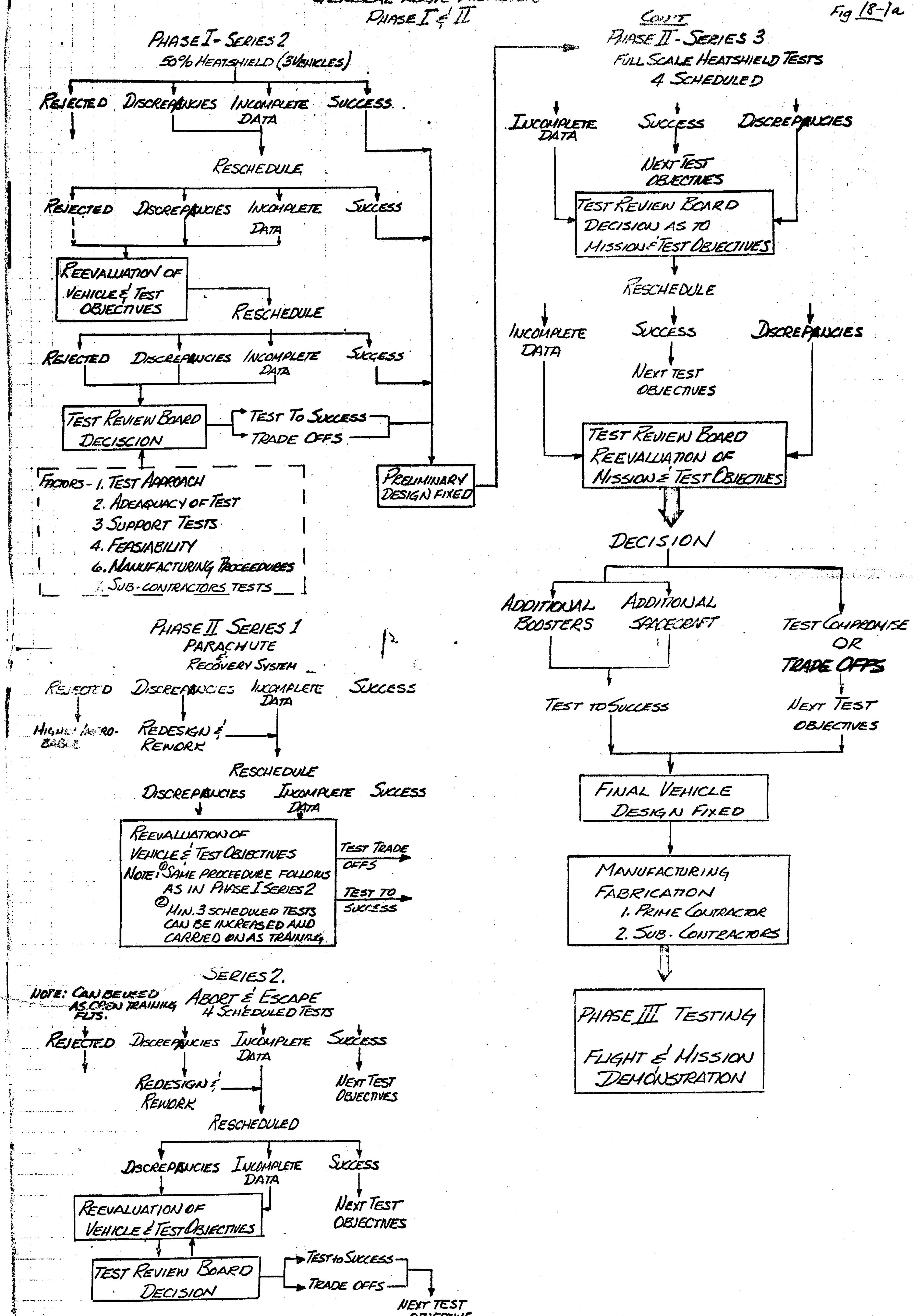


Fig 18-1b

Phase III – Flight & Mission Demonstration
Series 1. Flight Demonstration

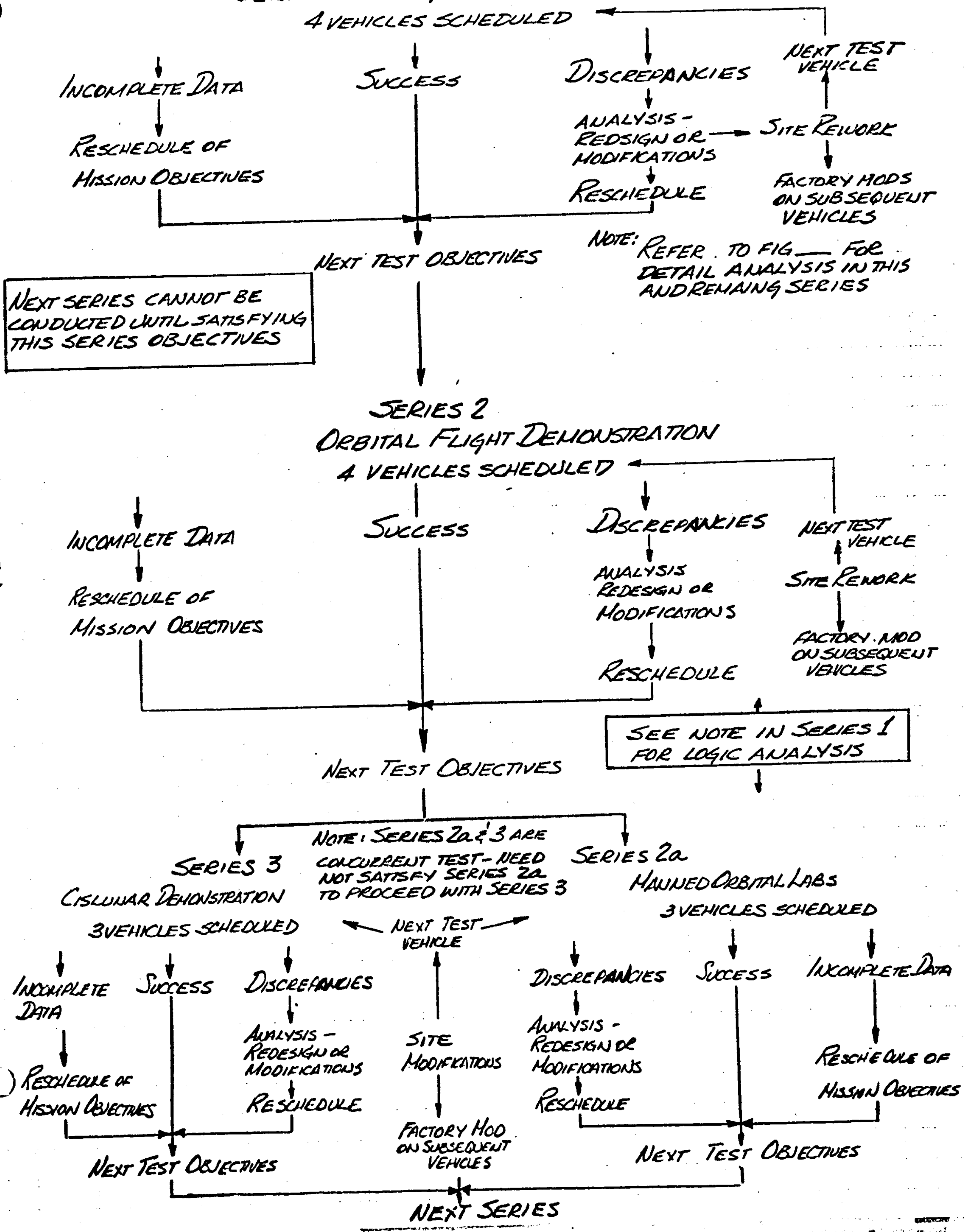
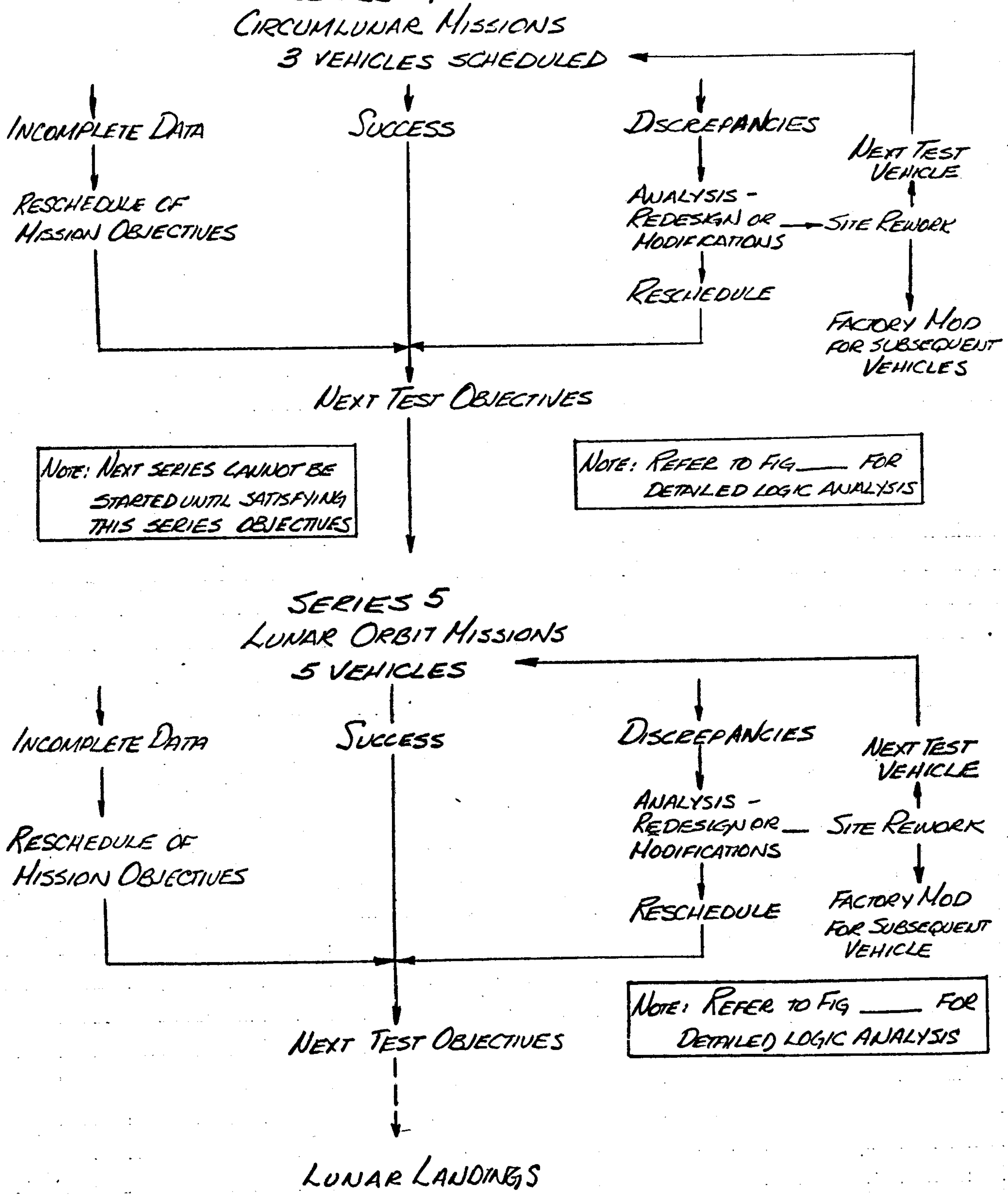


Fig 18-1c



SERIES 4
CIRCUMLUNAR MISSIONS
3 VEHICLES SCHEDULED
INCOMPLETE DATA
SUCCESS
DISCREPANCIES
NEXT TEST VEHICLE
RESCHEDULE OF MISSION OBJECTIVES
ANALYSIS - REDESIGN OR MODIFICATIONS
SITE REWORK
RESCHEDULE
FACTORY MOD FOR SUBSEQUENT VEHICLES
NEXT TEST OBJECTIVES
NOTE: NEXT SERIES CANNOT BE STARTED UNTIL SATISFYING THIS SERIES OBJECTIVES
NOTE: REFER TO FIG ___ FOR DETAILED LOGIC ANALYSIS
SERIES 5
LUNAR ORBIT MISSIONS
5 VEHICLES
INCOMPLETE DATA
SUCCESS
DISCREPANCIES
NEXT TEST VEHICLE
ANALYSIS - REDESIGN OR MODIFICATIONS
SITE REWORK
RESCHEDULE OF MISSION OBJECTIVES
RESCHEDULE
FACTORY MOD FOR SUBSEQUENT VEHICLE
NOTE: REFER TO FIG ___ FOR DETAILED LOGIC ANALYSIS
NEXT TEST OBJECTIVES
LUNAR LANDINGS

3/7/61

PLAN C – PHASE III Flight & Mission Demonstration

SERIES I Test a. UNDERSHOOT OR OVERSHOOT TRAJECTORY MANNED OR UNMANNED DEPENDING UPON PHASE II SERIES 3 RESULTS.

b. NOMINAL REENTRY – TRAINING

c. MANNED REENTRY – CREW TRAINING

d. MISSION ABORT, MECHANICS DEMONSTRATION

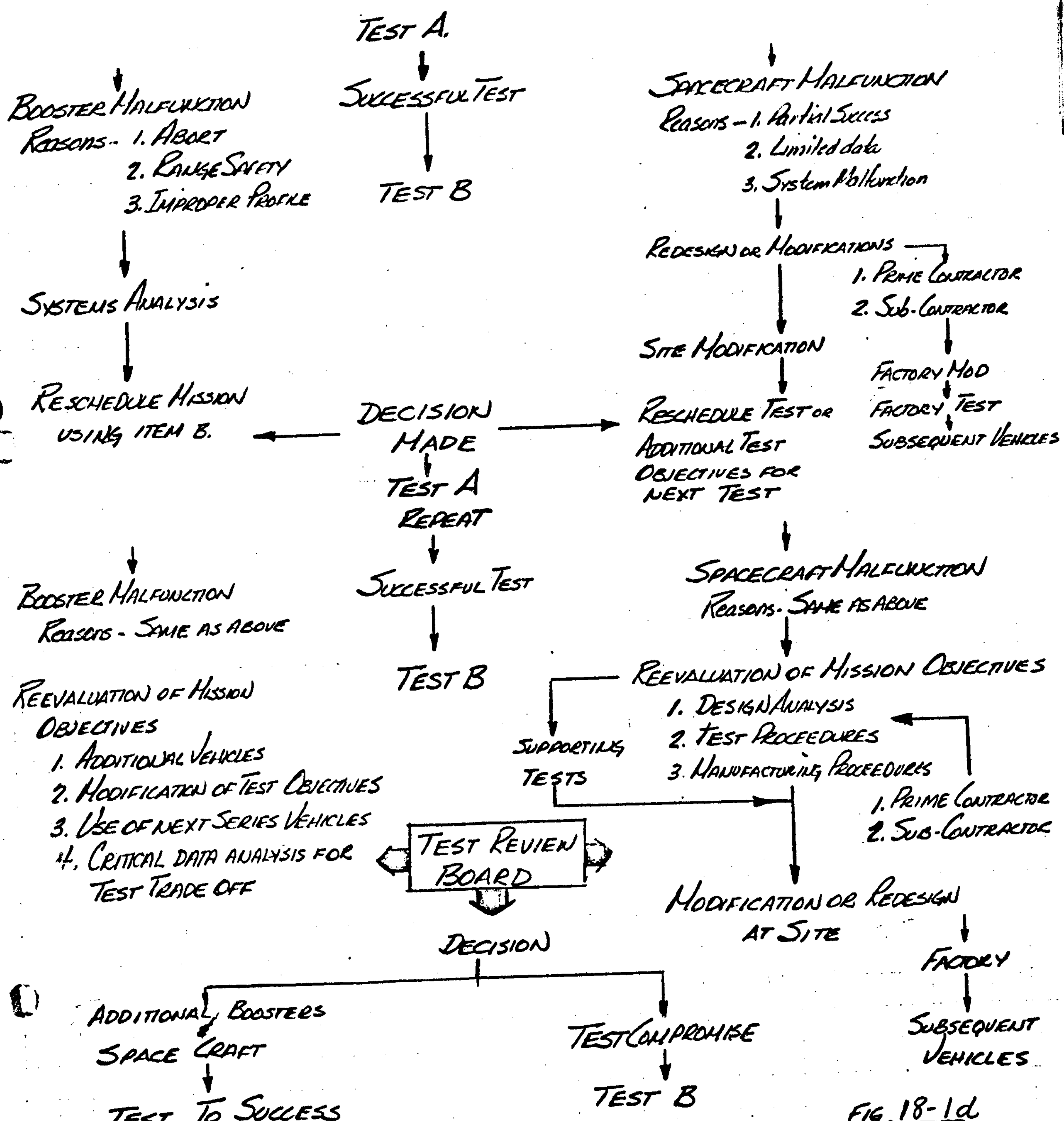


FIG 18-1d

[redacted]

INFLUENCE OF SPACECRAFT AND BOOSTER RELIABILITY ON SELECTION OF THE NUMBER OF VEHICLES FOR THE FLIGHT PROGRAM

Technical Memorandum 18-2

6 March 1961

R. Schwab

[redacted]

TM-18-2

INFLUENCE OF SPACECRAFT AND BOOSTER RELIABILITY ON SELECTION OF THE NUMBER OF VEHICLES FOR THE FLIGHT PROGRAM

Selection of Boosters and Spacecraft to carry out each Apollo mission has been made by examining the estimated reliability for each spacecraft and booster. Figure 18-2a relates spacecraft, booster, recovery and flight reliability with vehicles required, successful flights required and the probability of success of any particular flight series. The reliability numbers represent the middle ground between the optimistic and pessimistic prognostication. It is assumed that reliability is constant during a flight series utilizing similar boosters and spacecraft. In reality some improvement can be expected between the first and last flights of any series.

The constraints imposed in developing the flight test program are as follows:

1. Program completion date (first successful Lunar orbit).
2. Rate of Saturn launching (series connected flights for R/D program).
3. Booster, spacecraft, recovery data acquisition reliability.
4. Predicted availability of Saturn boosters, particularly R/D and early operational vehicles.
5. Probability of success of any flight series will be at least 85%.

The flight test plan was developed by first establishing the elements of the program: parabolic reentry, abort, earth orbit, etc. The overall program was then "roughed out", taking into consideration a completion date of lunar orbit in 1969, maximum launch rate of 4 per year, the span time to develop the Apollo hardware, and estimated booster availability reflected by current Saturn planning. Critical program phases and series were then pointed up by examining the chart in figure 18-2a. The first critical flight series occurs in Phase III, Series 3, Heat Shield demonstration flights using Saturn C-1 R/D boosters. Two successful flights are required before the program can continue: overshoot and undershoot reentry. Six flights must be scheduled to yield 85% probability of success of the series. Four Saturn C-1 R/D boosters are predicted to be available. Two Saturn C-1 operational boosters could be allocated to give a total of 6 vehicles. However, it is felt highly desirable to produce a manned Apollo flight as early as possible by using the first C-1 operational booster available. Therefore, this flight series is approached as follows:

1. Plan 4 flights - probability of success is 64%.
2. If first flight is successful, one success out of 3 remaining planned flights will yield about 80% probability of success. Continue program.
3. If first flight is unsuccessful, two successes out of 3 remaining yields a lowprobability of success of 53%. At this time, an alternate plan will be initiated to allocate 2 additional boosters and spacecraft to the program. Saturn C-1 operational boosters originally planned for Phase III, series 1 would be reassigned to Phase III, series 3. This allows a 21 month lead time for the replacement Saturn C-1 operational vehicles to be ordered and prepared.

[redacted]

This approach is to be followed in the Phase III flights where probability of success is lower than .85. The price paid in improving probability of success is basically program schedule. Assuming a leveling-off reliability for booster and spacecraft and a minimum launch rate (4 per year for Apollo-Saturn C-2), probability of success in a flight series can be improved by adding boosters and spacecraft to the program or by reducing the number of successful missions required in a series. The latter approach tends to reduce the reliability (probability of successful flight) because a number of objectives are crowded into one flight or insufficient number of parameters are investigated. Increasing the number of boosters and Spacecraft per series provides higher success probability at the expense of a later completion date since Apollo is a series stepped research and development program. The exception to this situation is the manned orbital laboratory flights which can continue at a more rapid launch rate and in any desired quantity while the lunar missions proceed.

Methods of flight program improvement can be illustrated by examining figures 18-2b and 18-2c which are the basis of establishing the number of vehicles required to meet a given probability of success. Three approaches offer higher probability of success:

1. Increase reliability of booster and spacecraft.
2. Increase the number of boosters and spacecraft.
3. Decrease the number of successful flights required.

The following example illustrates relative improvements:

Number of successful flights required = 3

Number of vehicles assigned to series = 5

Total booster - spacecraft reliability = 60%

Probability of success of series of 5 flights = 69%

[redacted]

$$\frac{\text{\% Improvement in Success Probability}}{\text{\% improvement in reliability}} = 1.6$$

Increase in success probability for additional flight = 14.5%

Increase in success probability for one less successful flight required = 32%

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PHASE	SERIES	SPACECRAFT	BOOSTER	Rs	Rb	Rr	Rd	Rt	Ns	Na	Nr .85	Nr .80	Nr .70	nr = Ns-Nr .85	.80	.70
I	1b	Re-entry Model	Altas-Agenda B	.85	.90	.95	.95	.69	2	4	4	3	2.5	0	0	H
II	1	Command Module	A/C	.99	.995	.98	.98	.95	2	3	3	3	3	0	0	0
	2	Command Module	Solid Booster	.95	.95	.95	.95	.77	3	5	5	4^+	4^+	0	+	+1
	3	Command Module	C-1 R/D	.90	.60	.90	.95	.46	2	4	6	5	4	-2	-1	0
III	1	Command Module	C-1 (OP)	.87	.70	.98	.95	.57	2	4	5	4	3	-1	0	+1
	2	Command & Mission	C-1 (OP)	.87	.75	.98	.95	.61	2	4	4	3^+	3	0	+1+	1
	2a	Command & Mission	C-1 (OP)	.87	.75	.98	.97	.62	2	3	4	3^+	3	-1	-1+	0
	3	Command & Mission	C-2 (OP)	.87	.70	.98	.93	.56	2	3	4	4	3^+	-1+	-1	+1+
	4	Command & Mission	C-2 (OP)	.85	.70	.98	.94	.55	2	3	5	4	3	-2	-1	0
	5	Command & Mission	C-2 (OP)	.86	.70	.98	.95	.56	3	5	7	6	5	-2	-1	0

Rs = Spacecraft Reliability
Rb = Booster Reliability
Rr = Recovery Reliability
Rd = Flight data return reliability
Rt = Total Flight Reliability
Ns = Number of successful missions required in series
Na = Number of flights assigned (Plan C) by program
Nr = Number of flights required for 85% probability of success per series
.80 = 80% probability of success
.70 = 70% probability of success

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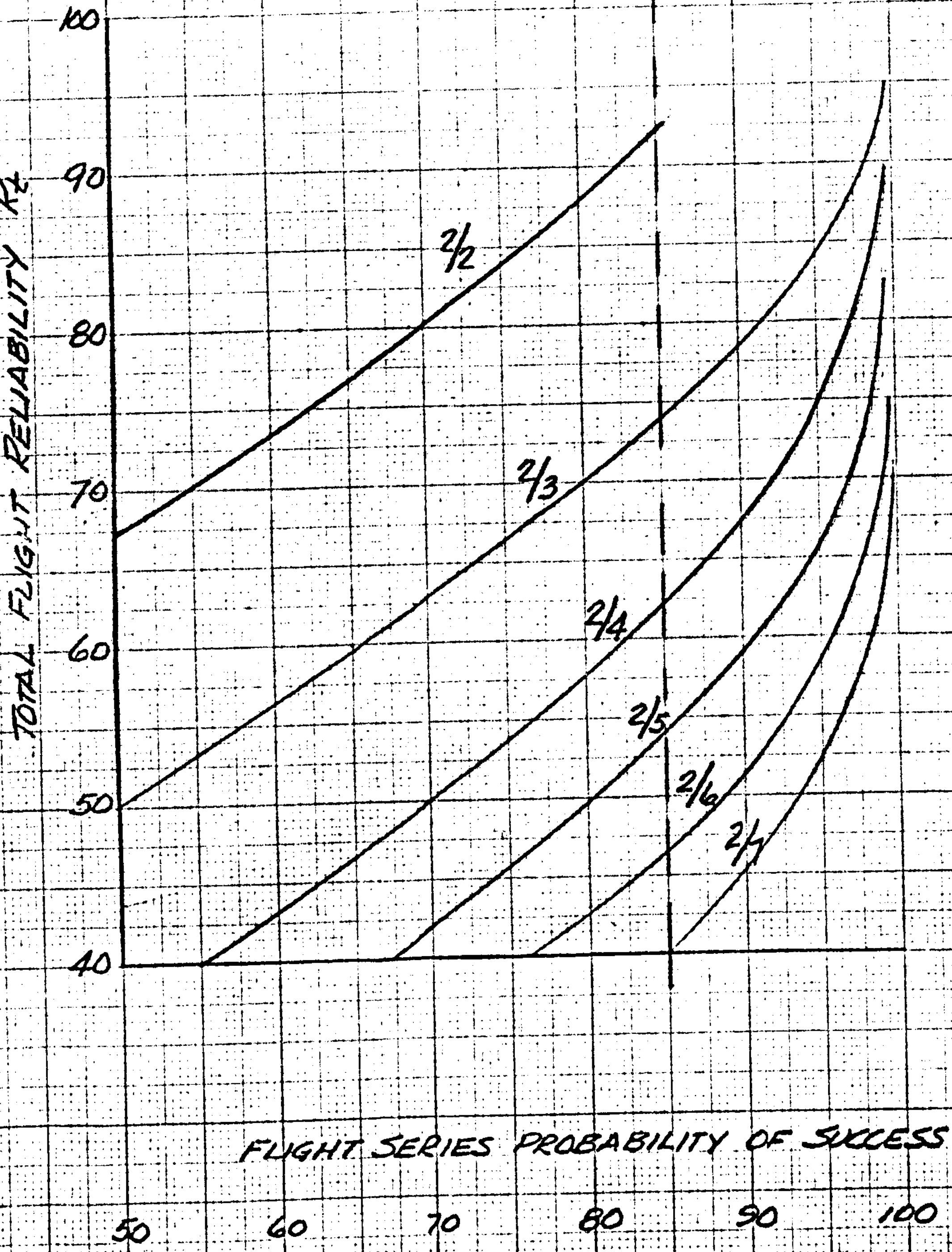
Fig. 18-2a Booster and Spacecraft Requirements based on estimated reliability.

[redacted]

RELIABILITY CONSIDERATIONS FOR DETERMINING SPACECRAFT & BOOSTER TEST QUANITIES FOR FIXED SUCCESSES vs VARIOUS TRIES

NOTE: 2/3 = SUCCESS/TRIES



TMIE —

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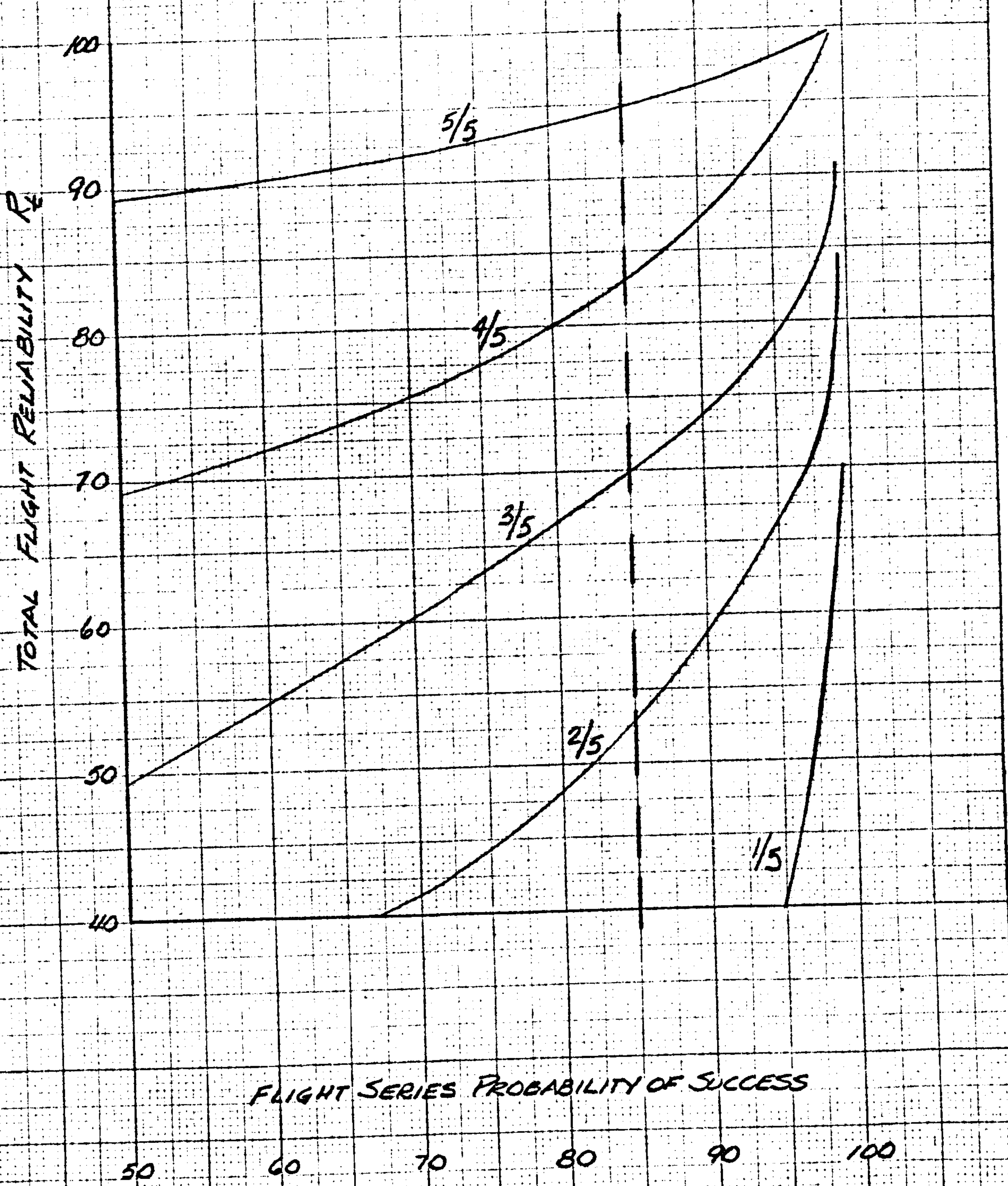
RELIABILITY CONSIDERATIONS FOR DETERMINING SPACECRAFT, BOOSTER TEST QUANITIES FOR VARYING SUCCESSES VS FIXED TRIES

NOTE: 4/5 = SUCCESS/TRIES



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[redacted]

A P O L L O

Mid-Term

C R E W T R A I N I N G

Technical Memorandum TM 18-3

THE MARTIN COMPANY

Baltimore 3, Maryland

6 March, 1961

R. Schwab
C. Brust

[redacted]

 [redacted]

The Apollo Program, with its crew concept flight vehicle, changes the complexity of the man's training and its philosophies in that man can (i.e. crew) assumes a more active participation in the flight. We must insure that his training encompasses all phases of system and vehicle function and malfunction and that through this he obtains the highest degree of confidence both in himself and his equipment. To accomplish this, it then becomes necessary that the crew personnel be a member of the contractors test organization, participating in design manufacturing and testing of these vehicles. Quite naturally, the contractor has neither the capability nor experience to fully satisfy all training; therefore, we must classify this training into two categories; Physiological Environment and Systems.

1. TRAINING AREAS

a. PHYSIOLOGICAL TRAINING -

1. NASA's responsibility for obvious reasons. Martin will furnish system requirement inputs where necessary i.e. flight data g's velocity, special orientation.

2. Existing trainers and training aids will require modification to Apollo parameters.

3. Cross training of crew members.

b. SYSTEM TRAINING -

1. Martin responsibility/NASA monitor

2. Build and maintain flight simulators

3. Train crew members in these simulators. This training, based on analysis of test program and system evaluation, to be varied to coincide with actual flight test results, where possible.

4. Phases of training -

Booster Flight

Coast

[redacted]

Injection

Abort - Mission and booster all phases

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#4. CONTINUED

Lunar Mission

Re-entry

Landing

1a. GROUND TRAINING

a. It is strongly recommended that ground training where it applies to system training be conducted at the contractor's facility. This aids in reducing the number of simulators required and shortens the communication link. It also tends to centralize equipment and personnel.

b. The methodoligy to be employed will follow that of previous programs performed by the Martin Company and will be the responsibility of the logistics support group.

With the crew members being a part of the test organization, they will have a closer relationship with the vehicle and systems thereby neatly assisting them in its understanding.

1b. FLIGHT TRAINING -

The program design is such that each flight is in itself a training mission and as such we need not designate specific flights with the prime objective of training.

a. AIRCRAFT - This is an area requiring further study - however, configure an A/C (Dual Capacity) to have flight characteristics with similar L/D as spacecraft displays and controls to be similar or same where possible. Primary objective is characteristics orientation.

orientation

b. PROTOTYPE TESTING (APOLLO) - Full scale free fall orientation. This phase would be either in later phases of testing in this series or as a carry on for training. Command module will have control systems and be proven before training flights.

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c. It is conceivable that every crew member will have had space orientation - Orbital or parabolic flight. This then will result in a ratio of space qualified crew members i.e. 2-1 or 1-2 crew alignment.

d. Should weightlessness present no major problems i.e., physical orientation), we then can possibly dispense with one of the prime objectives of the Mercury carry-on program.

e. Crew requirements for flight participation.

PHASE	SERIES	TEST	CREW	dtd.	REQUIREMENTS
III	1	a	A	2/65	a. Complete ground and flight training b. Mercury experience
		b	B	6/65	Same as above
		c	C	9/65	Same as above
		d	D	11/65	Same as above
	2	a	E	2/66	a. Same as above
		b	A	5/66	Same as above
		c	A_1	8/66	Same as above
		d	B	11/66	Same as above
III	2a	a	B_1	2/67	a. Same as above
		b	C	6/67	Same as above
		c	D	9/67	Same as above
III	3	a	(2-1 $E+C_1$)	1/67	a. Same as above + Orbital training
		b	$A+C_1$	4/67	Same as above
		c	A_1+C_1	7/67	Same as above
III	4	a	B	10/67	a. Same as above
		b	E	1/68	Same as above
		c	B_1	3/68	Same as above

[redacted]

PHASE	SERIES	TEST	CREW	dtd.	REQUIREMENTS
III	5	a	C	6/68	a. Same as above
		b	C_1	9/68	Same as above
		c	A	12/68	Same as above
		d	A_1	2/69	Same as above
		e	B	5/69	Same as above

Meteorites

TM-20

[redacted]

METEORITE PENETRATION STUDY

PROJECT APOLLO

TECHNICAL MEMORANDUM 20

BY: J. Heindl

Structures Group

Date: March 8, 1961

THE MARTIN COMPANY

Baltimore, Md.

[redacted]

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SUMMARY

The results of a preliminary study of the meteorite penetration problem has been presented in the manned lunar vehicle feasibility study (ref. 1). Further and more detailed analysis of this problem has been made during the initial study phase for two proposed Apollo spacecraft - the M-1-1 and the L-2C. This analysis was based on the 1957 meteorite model of Whipple (ref. 2) and the penetration equation of Summers (ref. 4). The probability analysis was based on that developed by Naumann (ref. 3).

The probability of no penetration was determined for various critical components of the two spacecraft and then combined to obtain an overall probability for the entire spacecraft. It was found that both the M-1-1 and the L-2C have an overall probability of no penetration of 0.97 for a 14 day mission.

The analysis made thus far show the meteorite problem is a definite design consideration but one to which reasonable answers may be found if the desired goal of probability of no penetration is in the order of 0.95. All analyses for meteorite penetration are, of course, limited by the present knowledge of the meteorite environment and the hyper-velocity impact penetration mechanism.

A discussion of the future study effort is also given.

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METEORITE MODEL

The meteorite model that has been used for the study analysis is Whipple's 1957 model as given in Table I of ref. (2). This model defines the meteorite mass, size, density velocity and frequency of occurrence. The meteorite density is 0.05 grams per cubic centimeter based on Whipple's "dust ball" concept. The data from ref. (2) is summarized in Table 1 of this memo.

PROBABILITY ANALYSIS

The methods outlined by Naumann (ref.3) has been used to establish probability curves showing the visual magnitude of meteorite that must be protected against in relation to the desired probability of no penetration and the exposure. The exposure is a function of of the exposed surface area of the vehicle and the time of the mission in space. Figures Ia and Ib show the probability of one or more hits on our square meter in one day plotted against the meteorite visual magnitude from Fig. 5 of ref. (3). Using the data of Figure Ia and Ib, it was possible to derive the visual magnitude of meteorite to be protected against for a given exposed area and time of exposure as shown in Figure 2. Knowing the visual magnitude, the size and velocity of the meteorite can be found by using the Whipple meteorite model - Shown in Table 1.

PENETRATION EQUATION

In order to determine the required skin thickness needed to resist penetration by a given meteorite, a penetration equation is required. For the study analysis, Summers' equation (ref. 4) has been used. This equation was selected because it shows a satisfactory correlation with impact test data over a wide range of velocities (up to 32,000 ft./sec.) and a range of projectile to target densities as shown in Figure 3. Summers' equation gives the penetration of a given diameter projectile into a thick target. The required thin single layer thickness was taken as twice the depth of penetration in a thick target given by Summers' equation. This is reasonable in that it has been found that a projectile can completely penetrate a target whose thickness is roughly one and one-half times as large as the penetration into a quasi-infinite target as stated in ref. (5).

[redacted]

DETERMINATION OF REQUIRED SKIN THICKNESS

Figure 4 shows the required skin thickness for various materials to resist penetration by meteorites of various magnitudes based on a meteorite density of 0.05 grams per cubic centimeter. Figure 5 shows a weight comparison for various structural materials. By the use of Figure 2 and Figure 4, the required skin thickness for various probabilities of no penetration, exposed surface area and time of exposure can be readily determined as shown in Figure 6 for aluminum alloy.

"BUMPER" CONCEPT

Preliminary studies using the curves of Figure 6 indicated a rather heavy gauge of aluminum (0.100" or more) would be required to obtain a probability of no penetration of 0.95 for the larger components of the Apollo spacecraft (e.g. the mission module). However, Whipple had proposed the meteorite "bumper" concept which indicated a significant reduction in total skin thickness by use of two skins suitably spaced apart.

[redacted]

"BUMPER" CONCEPT

From tests run by Olshaker (ref. 6), it appears that a "bumper" of suitable design may allow a reduction in a single layer thickness to one-third. Olshaker's data was obtained from tests using lead. However, some "bumper" test data using aluminum alloy is in general agreement with the lead test data. The Martin Company has made some "bumper" specimens of aluminum alloy. These specimens will be impacted with hypervelocity projectiles at the Ballistic Research Laboratory of the Aberdeen Proving Grounds. Data from these tests will be used to determine the "bumper" effectivity in aluminum alloy 14 S-T6 material. For the study, it has been assumed that the use of a "bumper" will allow a reduction to one-third of a single skin thickness as discussed in the preceding section.

It might be mentioned that five specimens of heat shield materials (ceramics and ablators) will also be tested by BRL to determine penetration resistance.

PENETRATION PROBABILITY FOR APOLLO SPACECRAFT

The results of the meteorite penetration study for the M-1-1 and L-2C spacecraft are shown in Tables 3 and 4. It will be noted that to obtain an overall vehicle probability of no penetration the exposed surface area and effective skin thickness of various critical components has been used. The external skin and the flame shield for the rocket engine surround the components of the mission module and the propulsion systems. Advantage has been taken of this surrounding skin to serve as a meteorite "bumper" to protect the enclosed components. In order to arrive at the probability of no penetration for the various components, it was necessary to determine the equivalent thickness of aluminum where the materials used are different (e.g. the heat shield). This was done by finding the equivalent penetration resistance of the material in terms of aluminum. The "bumper" effect of the external skin and flame shield has been used to arrive at an effective skin thickness using a factor of 3 (see Page 6 above). The heat shield construction will also serve as a "bumper". The results in Tables 3 and 4 show that the M-1-1 vehicle, for a 14 day mission, will have an overall probability of no penetration of 0.978 and the L-2C will have 0.976. This probability is based on a conservative assumption of no earth shielding, no mutual shielding of the components by each other and penetration based on meteorite impacts normal to surface. The solar cells are shown to suffer only a very small damage (less than 0.02%) due to meteorite impacts.

FUTURE STUDY EFFORT

Future effort during the study contract will involve the analysis of the W-1 Apollo spacecraft for meteorite penetration using the methods outlined in this memorandum. The test data from the aluminum alloy bumper specimens will be evaluated to determine the effect of the "bumper" configuration. The data from the impact tests of the heat shield materials will be evaluated to determine their penetration resistance. Further analysis will be made of the heat shield design of the command module as to its ability to withstand meteorite impacts.

The results of the study of meteorite penetration will be presented in a final report at the end of the study contract.

LIST OF REFERENCES

Ref. (1) Martin Co. E.R. 11245M Part 1
"Manned Lunar Vehicle System" Vol. 1
Feasibility Study

(2) Whipple, F. C. "The Meteorite Risk to
Space Vehicles" ARS Paper 499-57

(3) Naumann, R. J. "Meteoric Effects On
Long Range and Orbital Vehicles"
ABMA Report No. DS-TN-94
Sept. 27, 1954

(4) Summers, J. L. "Investigation Of High Speed Impact:
Regions Of Impact And Impact At Oblique Angles"
NASA TN D-94 October, 1959

(5) NASA Memorandum 10-18-58L
"Effect Of Target Thickness On Cratering And
Penetration Of Projectiles Impacting At
Velocities To 13,000 Feet Per Second"
W. H. Kinard et al Dec. 1958 CONF.

(6) Olshaker, A. E. "An Experimental Investigation
In Lead Of The Whipple "Meteor Bumper"
"Hypervelocity Impact" Fourth Symposium
ARGC-TR-60-39 September 1960

[redacted]

TM 20 - 10

TABLE I

WHIPPLE'S 1957 METEORITE MODEL

METEORITE DENSITY = .05 GMS/CM.^{3}

VISUAL MAG.	METEORITE MASS GRAMS	DIAMETER INCHES	VELOCITY FT./SEC.	NUMBER STRIKING SQUARE FOOT PER DAY
0	25.0	3.88	92,000	
1	9.95	2.86	"	
2	3.96	2.10	"	
3	1.58	1.55	"	
4	0.628	1.14	"	
5	0.250	.835	"	8.78×10^{-8}
6	9.95×10^{-2}	.615	"	2.56×10^{-7}
7	3.96×10^{-2}	.453	"	6.46×10^{-7}
8	1.58×10^{-2}	.333	88,600	1.62×10^{-6}
9	6.28×10^{-3}	.246	85,200	4.06×10^{-6}
10	2.50×10^{-3}	.181	82,000	1.022×10^{-5}
11	9.95×10^{-4}	.133	78,900	2.56×10^{-5}
12	3.96×10^{-4}	.098	75,800	6.46×10^{-5}
13	1.58×10^{-4}	.072	72,100	1.62×10^{-4}
14	6.28×10^{-5}	.053	69,000	4.06×10^{-4}
15	2.50×10^{-5}	.039	65,800	1.022×10^{-3}
16	9.95×10^{-6}	.029	62,300	2.56×10^{-3}
17	3.96×10^{-6}	.021	59,100	6.46×10^{-3}
18	1.58×10^{-6}	.016	55,900	1.62×10^{-2}
19	6.28×10^{-7}	.0114	52,800	4.06×10^{-2}
20	2.50×10^{-7}	.0084	49,300	1.022×10^{-1}
21	9.95×10^{-8}	.00615	"	2.56×10^{-1}
22	3.96×10^{-8}	.00452	"	6.46×10^{-1}
23	1.58×10^{-8}	.00314	"	1.62×10^{0}
24	6.28×10^{-9}	.00198	"	4.06×10^{0}
25	2.50×10^{-9}	.00125	"	1.022×10^{1}
26	9.95×10^{-10}	.00079	"	2.56×10^{1}
27	3.96×10^{-10}	.000497	"	6.46×10^{1}
28	1.58×10^{-10}	.000314	"	1.62×10^{2}
29	6.28×10^{-11}	.000198	"	4.06×10^{2}
30	2.50×10^{-11}	.000124	"	1.022×10^{3}
31	9.95×10^{-12}	.000079	"	2.56×10^{3}

1# = 453.6 GMS.

[redacted]

REF: TABLE I OF REF.(2)

J. HEINDL
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TABLE 2
METEORITE PENETRATION PROBABILITY - M-1-1 SPACECRAFT
14 DAY MISSION

COMPONENT	Equivalent Aluminum Skin Thickness	Effective Aluminum Skin Thickness With Bumper Effect	Surface Area FT.^{2}	Exposure FT.^{2}-Days	Probability Of No Penetration		
COMMAND MODULE	.063 "[1]	.309 "[2]	350	4900	.998		0.9784[10] OVERALL PROBABILITY
MISSION	.040	.240[2]	247.5	3460	.995		
MAIN PROPULSION SYSTEM HYDROGEN TANK OXYGEN TANK ENGINE	 .040 .040 .077 [4]	 .240[2] .390[2] .077	 317.0 85.3 16.6	 4440 1198 232	 .994 .9996 .992	.9857	
VERNIER ROCKET SYSTEM N_2H_4 TANK N_2O_2 TANK	 .070 .070	 .330[2] .330[2]	 19.6 19.6	 274 274	 .9998 .9998	.9996	
SOLAR CELL ARRAY	.020[7]	.020	288	4040	.99983[8]		
HELIUM RESERVOIR	.490[3]	1.59	5.50	77	>.9999		
EXTERNAL SKIN FLAME SHIELD	.040[5] .039[6]	.040 .039	625 121	8750 1692	<.005[9] 0.72[9]		

NOTES: FOR NUMBERED NOTES SEE PAGE 13 & 14

[redacted]

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TABLE 3
METEORITE PENETRATION PROBABILITY - L-20 SPACECRAFT
14 DAY MISSION

COMPONENT	Equivalent Aluminum Skin Thickness	Effective Aluminum Skin Thickness With Bumper Effect	Surface Area FT.^{2}	Exposure FT.^{2}-DAYS	Probability of no Penetration		
COMMAND MODULE	.060 * [1]	.300 * [2]	357.0	5000	.996		
MISSION	.040	.240 [2]	247.5	3460	.995		
MAIN PROPULSION SYSTEM HYDROGEN TANK OXYGEN TANK ENGINE	.040 .040 .077 [4]	.240 [2] .390 [2] .077	317.0 55.3 16.6	4440 1198 232	.994 .9996 .992	.9857	.9764 [10] OVERALL PROBABILITY
VERNIER ROCKET SYSTEM N_2H_4 TANK N_2O_4 TANK	.070 .070	.330 [2] .330 [2]	19.6 19.6	274 274	.9998 .9998	.9996	
SOLAR CELL ARRAY	.020 [7]	.020	288	4040	.99983 [8]		
HELIUM RESERVOIR	.490 [3]	1.59	5.50	77	> .9999		
EXTERNAL SKIN FLAME SHIELD	.040 [5] .039 [6]	.040 .039	625 121	8750 1692	< .005 [9] 0.72 [9]		

NOTES: FOR NUMBERED NOTES SEE PAGE 13 & 14

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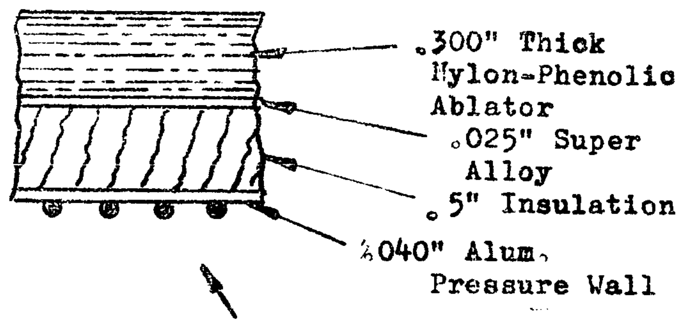
NOTES FOR TABLES 2 & 3

Notes:

1. M-1-1 Average Heat Shield Cross Section



L-2C Average Heat Shield Cross Section — — — — — — — — (Same as M-1-1 except Ablator is ¼")

2. Assuming bumper increases effective skin thickness by a factor of 3, i.e. effective skin = 3 x (Bumper thickness + Basic thickness).

3. 0.240" steel shell equivalent to 0.49" aluminum.

4. 0.040" steel cooling tubes equivalent to .077" aluminum.

5. 0.040" aluminum external skin acts as bumper.

6. 0.020" steel flame shield acts as .039" aluminum bumper.

7. 0.025" silicon cells equivalent to .020" aluminum.

8. Solar Cell Effective Area

    Cell size 3/8" x 3/4" .025 silicon face

    10,000 cells/array   12 arrays

    Total cells = 120,000 arranged in groups of five

    Visual magnitude meteorite to destroy .025" silicon face is 16.2 or less (estimated)

    Estimated number of hits for 14 day mission is four for visual magnitude 16.2 or less

    Assuming 4 hits destroy 4 banks of 5 cells the damage is

    $$\frac{4 \times 5}{120{,}000} = .000167 \; (.017\%)$$

    Effective area = 1 - .000167 = .99983

9. External skin and flame shield serve as meteorite bumper. Penetration is not a problem.

10. Overall probability of no penetration for no earth shielding, no mutual shielding of components and all meteorite impacts normal to surface.

[redacted]

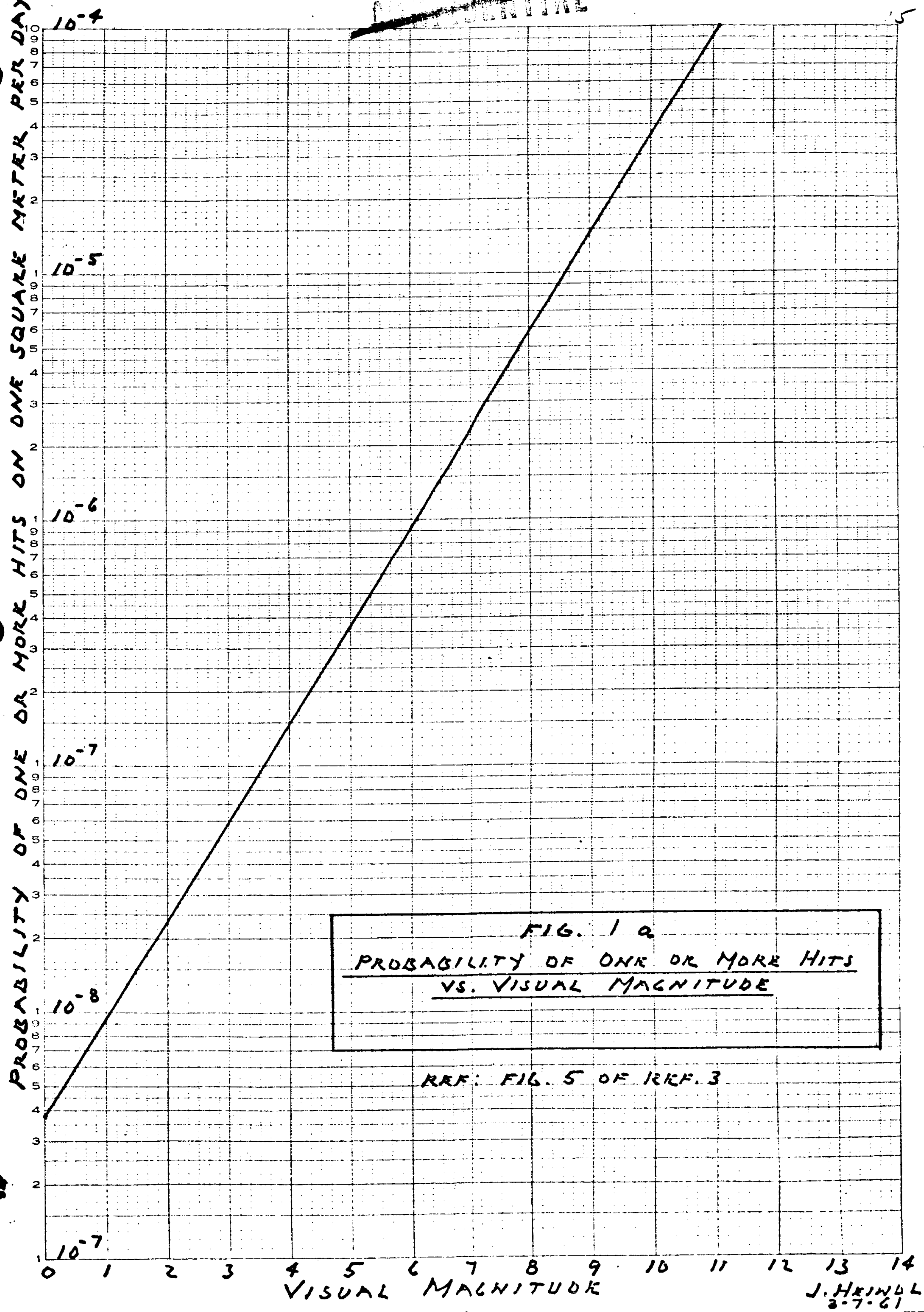


FIG. 1 a
PROBABILITY OF ONE OR MORE HITS VS. VISUAL MAGNITUDE

[redacted]

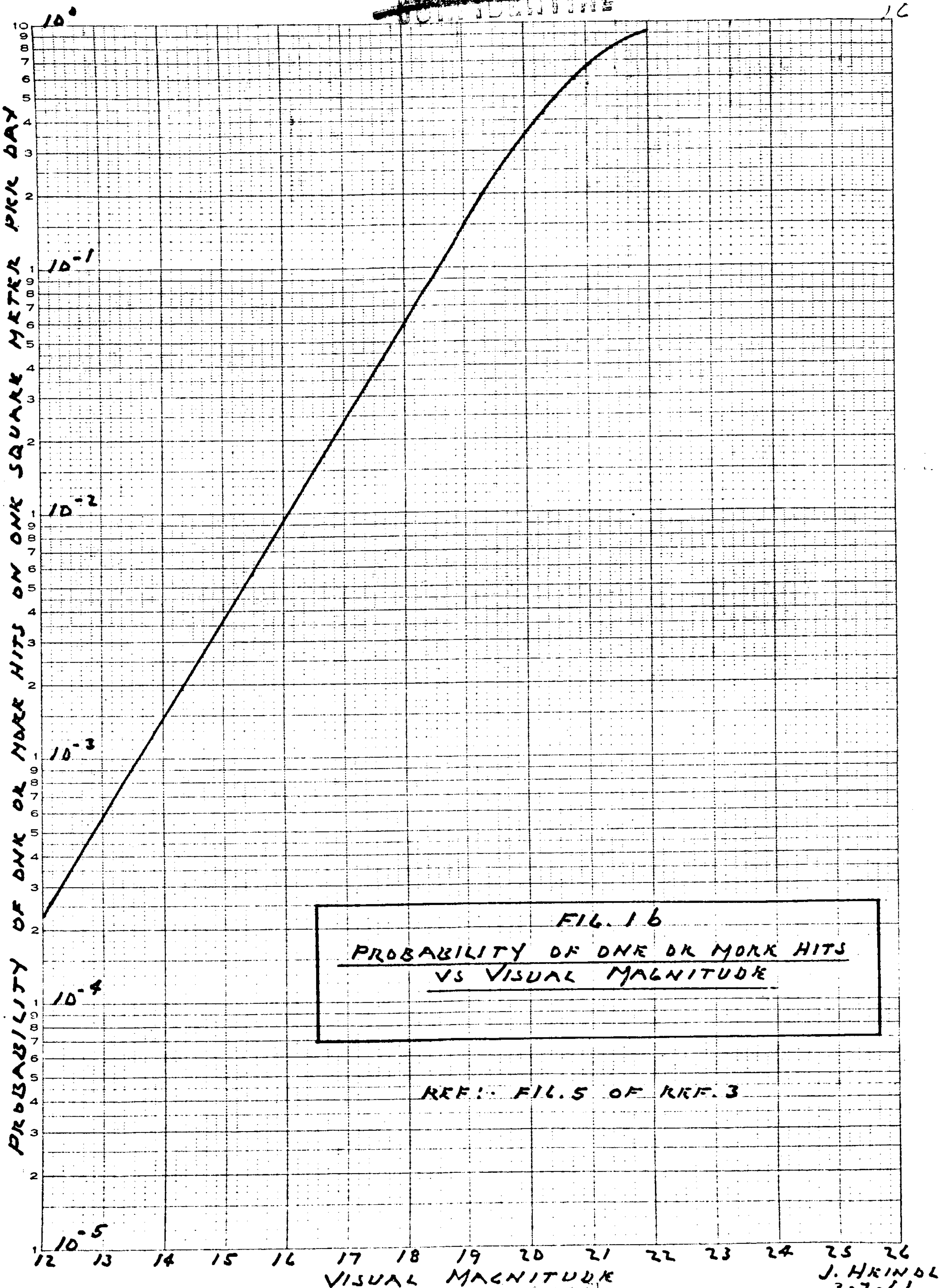


FIG. 16
PROBABILITY OF ONE OR MORE HITS VS VISUAL MAGNITUDE

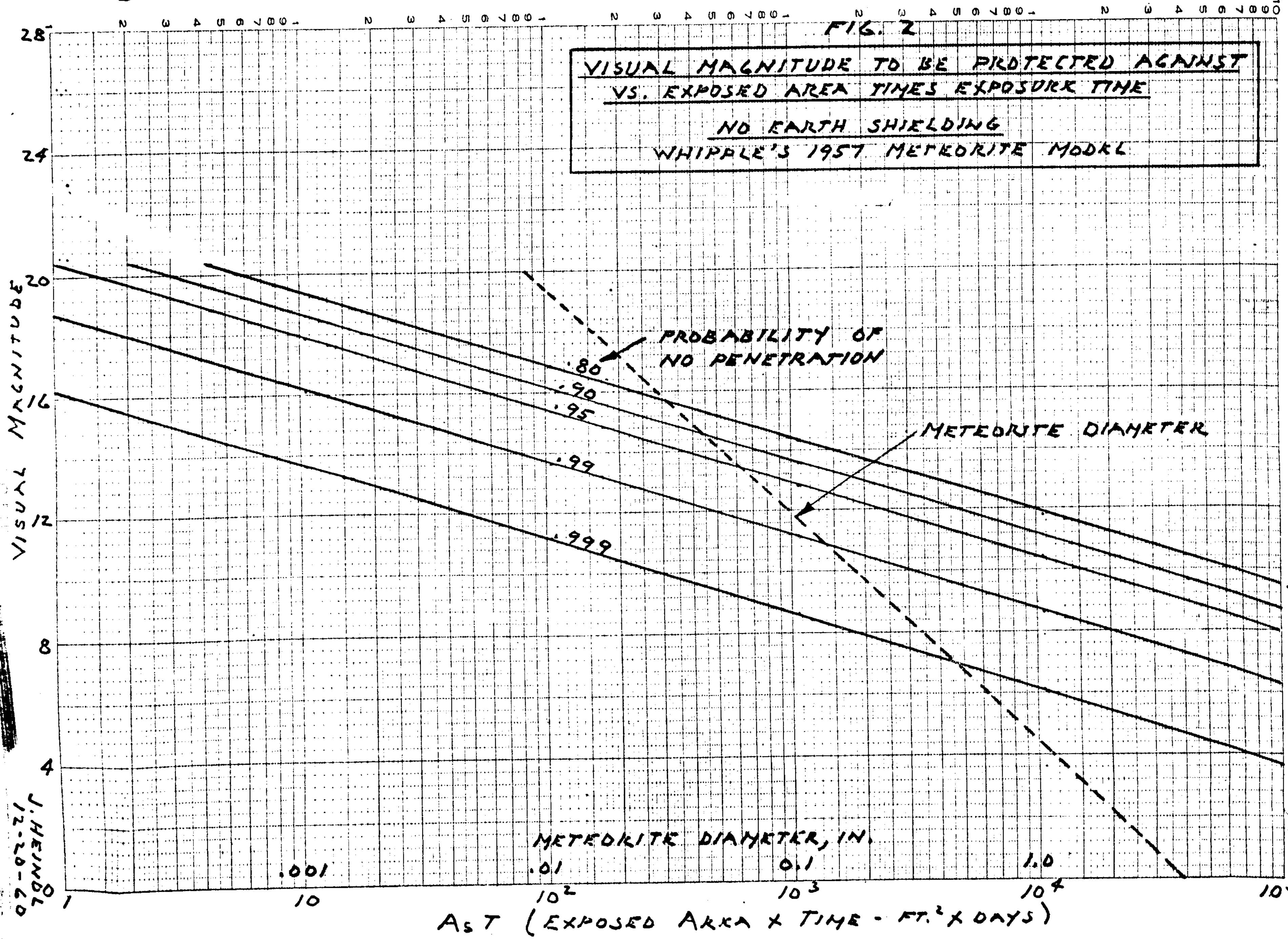


FIG. 2

VISUAL MAGNITUDE TO BE PROTECTED AGAINST VS. EXPOSED AREA TIMES EXPOSURE TIME

NO EARTH SHIELDING

WHIPPLE'S 1957 METEORITE MODEL

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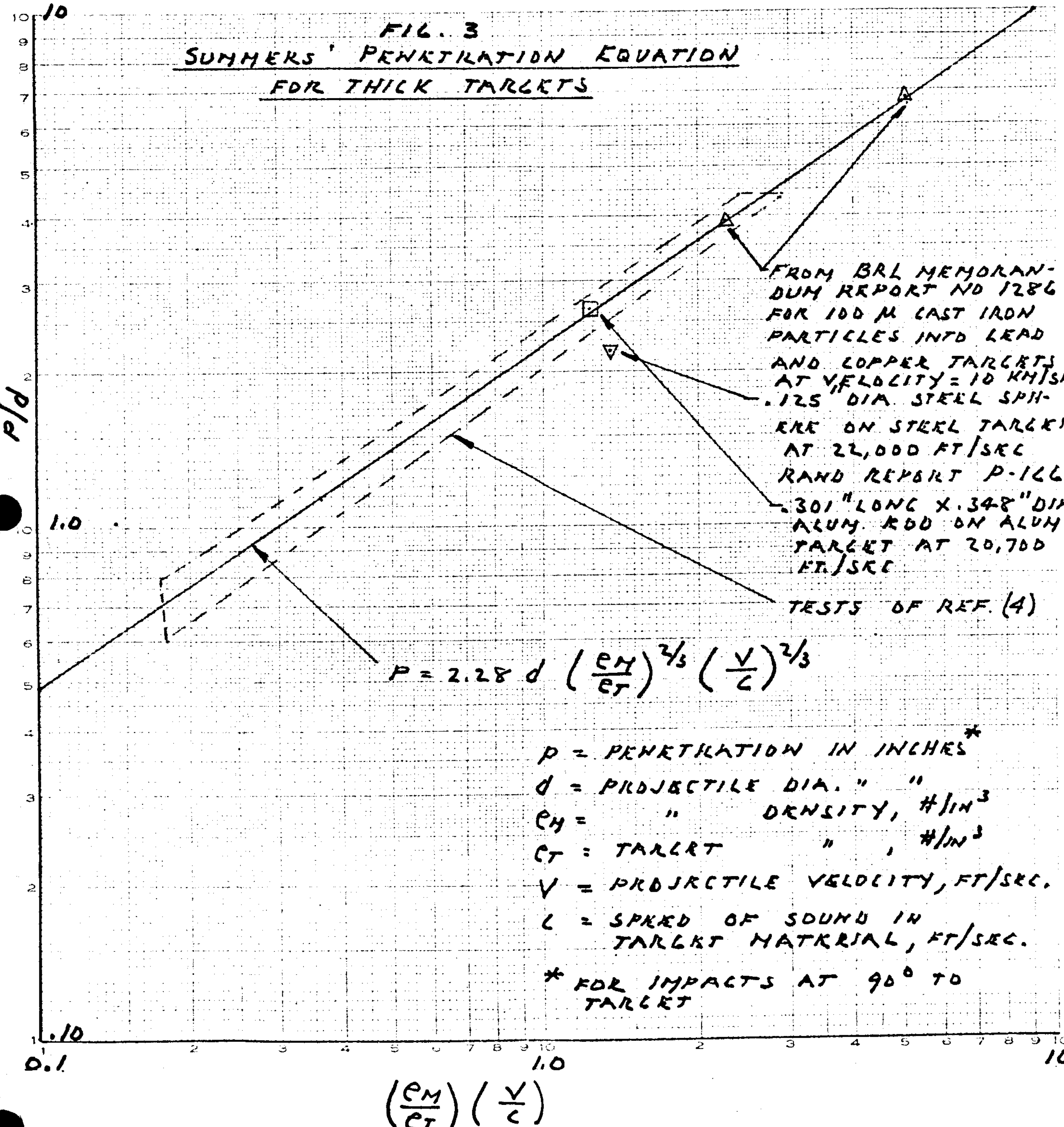
12-20-60

FIG. 3
SUMMERS' PENETRATION EQUATION FOR THICK TARGETS



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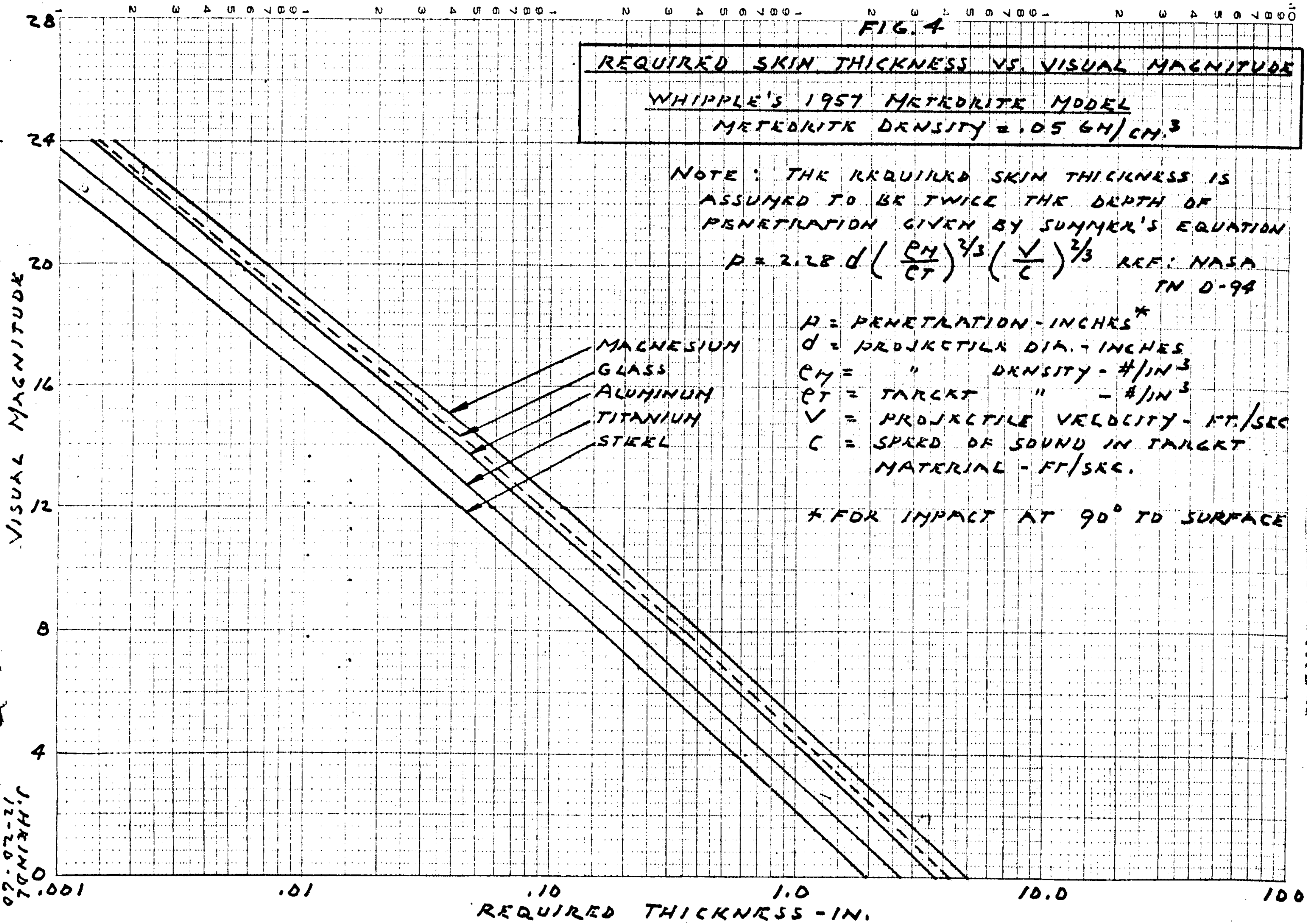

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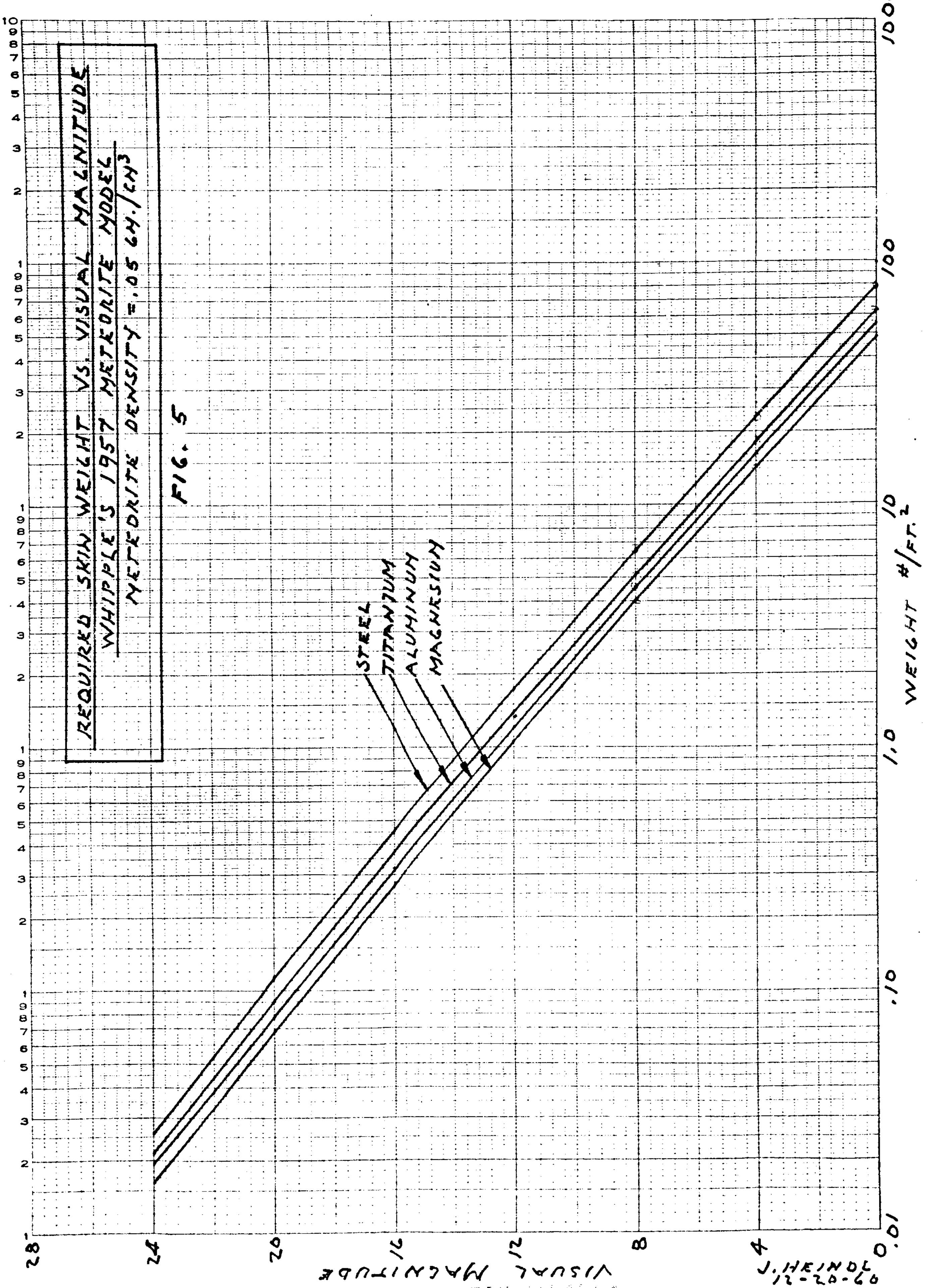
REQUIRED SKIN WEIGHT VS. VISUAL MAGNITUDE
WHIPPLE'S 1957 METEORITE MODEL
METEORITE DENSITY = .05 GM./CM³
STEEL
TITANIUM
ALUMINUM
MAGNESIUM
WEIGHT #/FT.²
1000
100
10
1.0
.10
0.01
VISUAL MAGNITUDE
28
24
20
16
12
8
4
0
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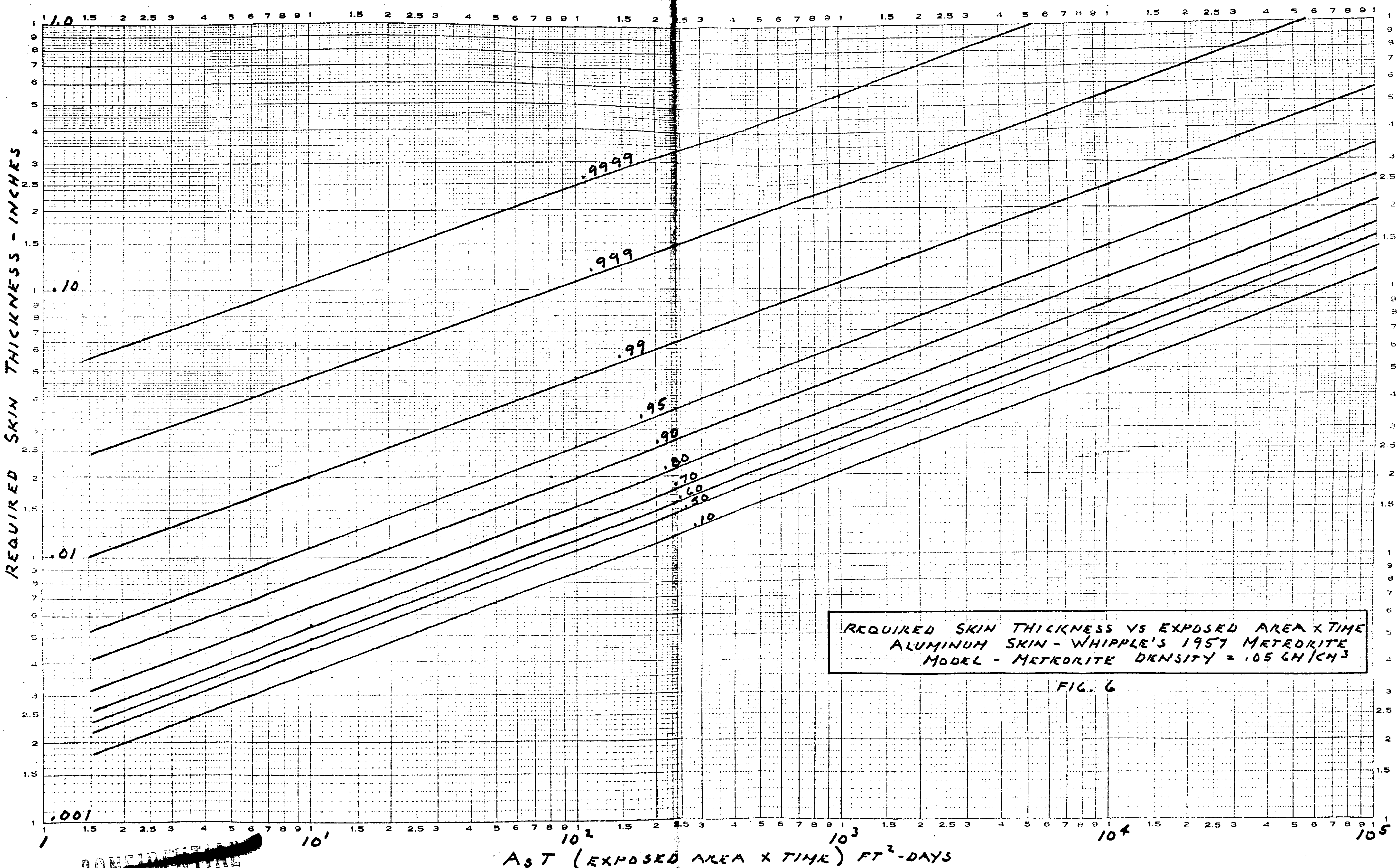
12-20-60

FIG. 5

[redacted]



REQUIRED SKIN THICKNESS VS EXPOSED AREA x TIME
ALUMINUM SKIN - WHIPPLE'S 1957 METEORITE
MODEL - METEORITE DENSITY = .05 GM/CM³

FIG. 6

[redacted]

J. HEINDL
3-7-61

Aero Control System (Hot Gas)

TM-21

[redacted]

3-7-61

TECHNICAL MEMO - 21

Aerodynamic Power Control System

In studying the design requirements for the reentry-aerodynamic power control system, it became increasingly apparent that the conventional hydraulic powered system has serious limitations. Closed cycle system leakage, fluid temperature limitations, and redundancy requirements to achieve reliability with the complex hydraulic system brought us to the realization that a hot gas system should be evaluated. It is believed that by proper design and component selection either system could be made to satisfy the functional system requirements and that system weight should be the ultimate basis for selection.

The results of the evaluation, fig. 1, fig. 2, of a conventional dual hydraulic system powered from H_2 and O_2 fuel A.P.U. turbines against both mono and bi-propellant hot gas systems have shown a decided weight advantage in favor of the hot gas system. The hot gas system has several other significant advantages: high ambient temperature compatibility, capability of using the gas source for both the aerodynamic and reaction control systems, and system simplicity through a single energy conversion from fuel to actuator.

The hot gas systems evaluated were complete systems while the hydraulic system included only incremental fuel and turbine weight increases over a system already installed in the vehicle. The weight difference would have been considerably larger if all the fuel system and turbines had been charged to the hydraulic system alone.

On the basis of this evaluation, a hot gas system, fig. 3, has been selected for Apollo. The selection of the fuel to be used will be made in the last half of the program after a more detailed study into the possible integration of fuels with other vehicle power systems.

[redacted]

Configuration		L2C							W-1						
Max. Dyn. Press.	Lb/ft^2	505							725						
Surface Areas	Ft^2	9.0							6.5 (Pitch & Roll) EA - 5.5 Yaw Ea.						
Hinge Moment	Ft-lbs	10,000							6640						
Reentry Time	Sec.	606							980						
Surface Defl.	Deg.	140							60						
Surface Rate	rad/sec	.5							.5						
System		Hot Gas						Hyd.	Hot Gas						Hyd.
System Fuel		H_2O_2		N_2H_4		H_2+O_2		APU	H_2O_2		N_2H_4		H_2+O_2		APU
System Temp.	°F	1400		1600		1600		450	1400		1600		1600		450
System Press.	PSI	600	1200	600	1200	600	1200	3000	600	1200	600	1200	600	1200	3000
Surface H.P. (E.A.)	H.P.	11.6	11.6	11.6	11.6	11.6	11.6	13.1	6.04	6.04	6.04	6.04	6.04	6.04	6.04
Hyd. Pump H.P. (Peak)	H.P.	-	-	-	-	-	-	54.5	-	-	-	-	-	-	33.4
Tot. System Wt.	Lbs.	235	213	224	203	222	201	349	241	216	224	202	220	199	339

Notes

1. Based on 40 N.M. corridor undershoot trajectories.
2. Hydraulic system fully redundant except actuator, APU fuel & gas gen.
3. Hyd. System wt. assumes vehicle has installed dual H_2 & O_2 driven APU turbines. Wt. increment added for increase in turbine & gear box size from 5 H.P.
4. Hot gas separate system. Redundant servo motors & valves.

TM-21

Fig. 1

[redacted]

Typical Weight Summary
W-1 Configuration

HOT GAS

Energy source - H_2O_2 fuel

System pressure - 1200 PSI

System temperature - 1400°F

Harmonic drives (4)	89.0
Pneumatic motors (8)	22.6
Servo valves (8)	10.0
Transmission lines	9.5
Gas generator, misx. valves, fuel and tankage	85.9
Total Weight	217.0

HYDRAULIC

Energy source - $H_2 + O_2$ Turbine APU's

System pressure - 3000 PSI

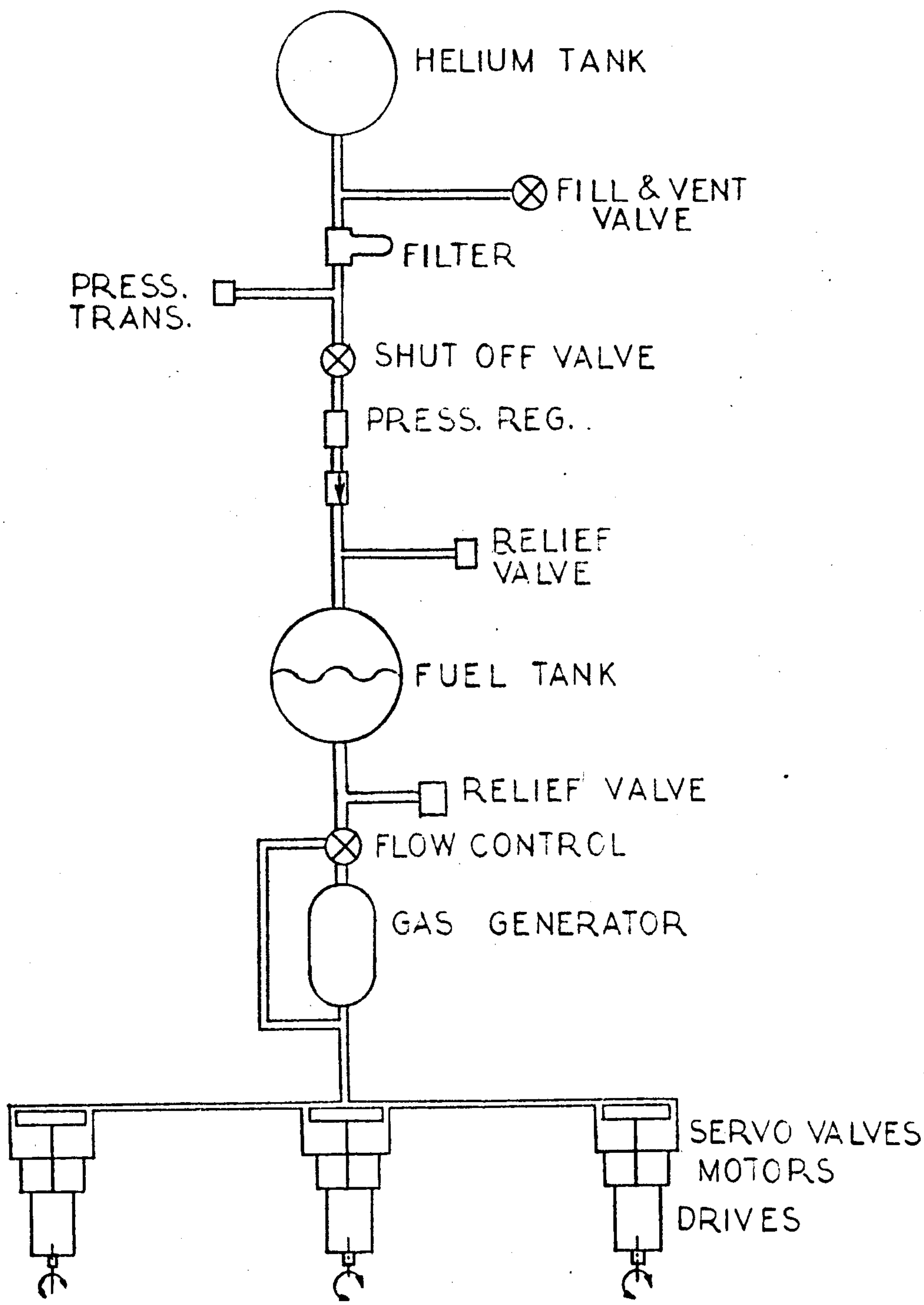
System temperature - 450°F

Tubing	42.0
Actuators (3)	72.0
Filters	12.0
Check Valves	3.0
Relief Valves	6.5
Servo Valves	8.0
Reservoirs	24.0
Pumps	30.0
Fluid	54.4
Cooling	20.0
Insulation	20.0
Fuel	7.4
APU Weight Increase	40.0
Total Weight	339.3 lbs.

TM-21
Fig. 2

[redacted]

AERO CONTROL POWER SYSTEM



TM-21
FIG 3

Weight Report

TM-22

APOLLO

Mid-Term

Weight and Balance Report

Technical Memorandum 22

The Martin Company

Baltimore, Maryland

March 2, 1961

I. INTRODUCTION

This report presents the results of the first half of the Apollo study of vehicles for circumlunar, lunar orbit, and lunar takeoff missions.

Analyses of ten re-entry bodies have been made in detail and the weight comparisons are shown in Tables I and II. From these data, four vehicles were selected for more careful analysis. The weight summaries of these four vehicles are shown in Table III. A brief description and a detail weight breakdown of the major components of the selected designs follow Table III.

During the second half of the Apollo study the weight of the selected configurations or configuration will be further defined and optimized. The center of gravity will be positioned properly. Requirements for detail design weight and center of gravity optimization will be determined. Specifications for actual weighing, actual C.G. determination and possible moment of inertia determination will be prepared.

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II SUMMARY

TABLE I

APOLLO

(A) Spacecraft Weight Comparison

	Modular						Integrated		Fly-In	
	W-1	M-1-1	L-2C	L-1	L-8	Flapped Mercury	W-1	L-2C	Lenticular	M-2
Command Module	5,847	5,958	5,711	5,856	5,929	5,742	7,749	7,982	6,707	8,850
Launch Escape System	1,000	1,000	1,097	1,000	1,000	1,097	1,280	1,498	1,260	1,470
Mission Module	4,345	4,417	4,271	4,300	4,220	4,337	2,606	2,597	4,624	4,441
Propulsion System and propellant	4,332	4,374	4,298	4,324	4,322	4,306	4,356	4,432	4,622	5,140
Circumlunar Total at Launch (effective)	14,724	14,949	14,577	14,680	14,671	14,682	15,191	15,709	16,413	19,101
△ Propellant and Equipment	4,084	4,144	4,034	4,072	4,070	4,047	4,040	4,137	4,465	5,125
Lunar Orbit Total at Launch (effective)	18,808	19,093	18,611	18,752	18,741	18,729	19,231	19,846	20,878	24,226
△ Propellant	3,349	3,400	3,305	3,338	3,337	3,317	3,377	3,460	3,678	4,248
Lunar Takeoff (effective)	22,157	22,493	21,916	22,090	22,078	22,046	22,608	23,306	24,556	28,474

TABLE II

APOLLO

(B) Re-entry Body Comparison

	W-1	M-1-1	L-2C	L-1	L-8	Flapped Mercury	W-1 Integrated	L-2C Integrated	Lenticular	M-2
Heat Shield	1140	1148	1108	1190	1180	1158	1335	1477	1176	1841
Heat Shield Water and System	180	180	172	218	187	185	201	209	193	267
Structure	920	1075	870	1007	854	912	1156	1160	1247	1693
Aerodynamic Surface or Flap	251	221	173	188	295	155	320	251	371	872
Surface Controls	258	236	290	155	315	234	330	403	425	600
Reaction Controls	160	160	160	160	160	160	205	225	185	244
Landing System	450	450	450	450	450	450	575	630	622	845
Auxiliary Power System	527	527	527	527	527	527	675	675	527	527
Environmental Control (P&E)	310	310	310	310	310	310	973	973	310	310
Instruments	260	260	260	260	260	260	260	260	260	260
Instrumentation	98	98	98	98	98	98	122	122	98	98
Communications	134	134	134	134	134	134	159	159	134	134
Guidance	220	220	220	220	220	220	220	220	220	220
Scientific Equipment	70	70	70	70	70	70	175	175	70	70
Furnishings and Equipment	239	239	239	239	239	239	413	413	239	239
Crew	630	630	630	630	630	630	630	630	630	630
Total Launch Weight of Re-entry Body	5847	5958	5711	5856	5929	5742	7749	7982	6707	8850

[illegible]

TABLE III

APOLLO

(C) Selected Spacecraft Comparison

	W-1	W-1 Integrated	L-2C	L-2C Integrated
Command Module	(5847)	(7749)	(5711)	(7982)
Heat Shield	1320	1536	1280	1686
Structure and Controls	1589	2011	1493	2039
Crew and Equipment	2938	4202	2938	4257
Launch Escape System	1000	1280	1097	1498
Mission Module	(4345)	(2606)	(4271)	(2597)
Crew Capsule and Equipment	1837	--	1837	--
Structure and Controls	1912	1660	1848	1661
Equipment and Systems	596	946	586	936
Propulsion System and Propellant	4332	4356	4298	4432
Total Circumlunar (Effective) *	14724	15191	14577	15709
Δ Propellant and Equipment	4084	4040	4034	4137
Total Lunar Orbit (Effective)	18808	19231	18611	19846
Δ Propellant	3349	3377	3305	3460
Total Lunar Take-Off (Effective)	22157	22608	21916	23306

* "effective" weight includes a reduction of 800 lb. to account for the fact that the launch escape tower is jettisoned at 300,000 ft. rather than carried to escape velocity.

[redacted]

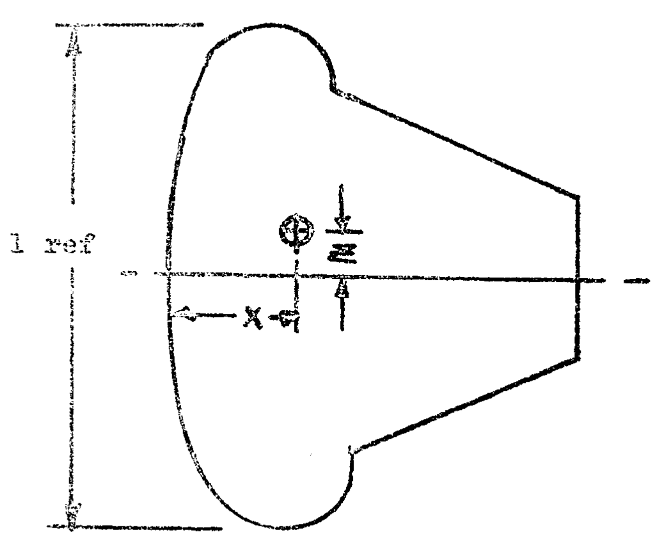
II SUMMARY

(D) CENTER OF GRAVITY

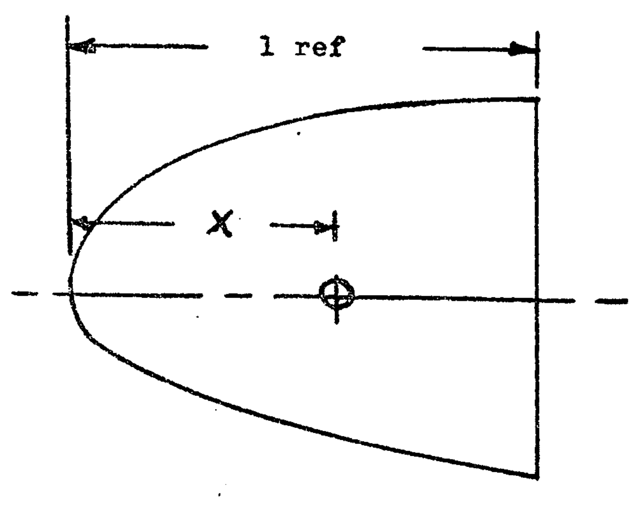
	L-2C	Mod. W-1	Integrated W-1
Most Aft Permissible Longitudinal C.G.(X)	20.0%	62%	62%
Longitudinal C.G. Calc. (Before Ablation)	19.5%	65.3%	60.6%
Longitudinal C.G. Calc. (After Ablation)	20.1%	68.3%	63.6%
Vertical C.G. Desired (Z)	3%	0	0
Vertical C.G. Calc. (Before Ablation)	1.4%	0	0
Vertical C.G. Calc. (After Ablation)	1.0%	0	0

C.G. in percent of 1 ref shown below

L-2C



W-1



III DETAIL ANALYSES OF 4 SELECTED SPACECRAFT

A LAUNCH ESCAPE SYSTEM

Thrust required to separate the Command Module from the Mission Module under abort conditions is critical at maximum Q for all configurations. The thrust required is 77,500 lbs. for 2 sec. for the W-1 and 120,000 lbs. for 1 sec. for the L-2C. This differences in thrust requirements is primarily due to the drag difference between these vehicles.

The same rocket is used in both cases except for a slight modification. The L-2C rocket is increased in length to reduce burning time to one second. A weight penalty of 50 lbs. is charged to the empty case for the additional length increment.

Tower length is established by CG requirements of the Command Module and rocket package on the L-2C. Minimum escape system length is 200 in. from the nose cap of the Command Module to the LE of rocket on the L-2C. On the W-1 the tower length is established by rocket blast clearance requirements. The CG requirements on this vehicle are met by separating the Command Module from the Mission Module at a frame 48" aft of the normal recovery separation frame. In this manner the skirt remains with the Command Module and moves the AC aft of the CG. When there is no emergency escape requirement, such as when re-entering from orbital flight, the separation between the Command Module and Mission Module takes place at the recovery separation frame.

Weights for the integrated configurations were approximated. A later review indicates the approximations shown below for the integrated versions are too heavy by 200 lb. in both cases.

	W-1	L-2C	W-1 Integ.	L-2C Integ.
Rocket				
Inert	186	236		
Propellant	664	664		
Tower	110	163		
Separation System	40	34		
Total	1000	1097	1280	1498

B COMMAND MODULE

1 Heat Shield

The heat shield is made up of a composite of ablators with either structural insulation and cooling, or a radiant type shield with insulation and structural cooling. The ablator is used to absorb the high heat rate peaks, the other underlying layers are used to resist the long time, lower heating rates.

The heat shield weight is based on consideration of an overshoot boundary at $C_{L\ Max}$ without skip and an undershoot corridor of 60 naut. mi. from the overshoot boundary. No lateral or local maneuvering has been considered at this time.

The heat shield design personnel have applied a 25% factor to the heating rate. The weight group has applied a nonoptimum factor of 15% to the basic heat-shield weight.

In the case of the vehicles selected for further study, the L-2C weight is based on trajectory data for that vehicle. The W-1 heat shield weight is based on the M-1 data since specific information was not available for the W-1. Some early indications are that this assumption may prove to be optimistic. The integrated vehicles had approximately the same wing (surface) loading as the modular concepts and therefore the same unit weight for shielding was maintained.

Pertinent data with regard to the weight breakdown are as follows:

(1) Density of nylon-phenolic ablator is 76 lb/cu ft.

(2) An allowance of 1 lb/sq ft is assumed for the weight of the super alloy metal and its installation.

(3) Density of ADL-17 insulation, which is used with the super alloy radiation shield is 12 lb/cu ft.

(4) Weight of the structural insulation is estimated at 1 lb/sq ft. This insulation is used when the insulation requirements are relatively low. Though not as effective as super alloy and ADL-17, there is no weight penalty required for attachment since this material can be bonded directly to the ablator and the pressure structure. Another advantage of the structural insulation is that the bond is good at higher temperatures than the super alloy bond and the ablator need not be used for insulation to keep the bond temperature at satisfactory levels.

(5) One-half of the insulation weight is subtracted from the shield weight and allotted to the shield-cooling-system weight.

Tables IV and V show the heat-shield weight distribution for the L-2C and the W-1 modular concepts.

The heat-shield weight summary is as follows:

	W-1	L-2C	W-1 Integ.	L-2C Integ.
Ablator	437	377	978	961
Super Alloy and Installment Wt (1 lb/□')	244	216		
Insulation (ADL-17 and Structural Insulation)*	126	117		
Shielding for Parachute and Rocket Containers	163	-	163	-
Prov. for Parachute Risers	20	-	20	-
Door Cut-Out	-	22	-	44
Window Assembly	-	30	-	30
Structure Added for Movable Shield at Landing	-	181	-	249
Nonoptimum Factor	150	145	174	193
Total Weight	1140	1188	1335	1477
Surface Area (Sq. Ft.)	343	340	407	461

* Insulation weight shown is only that part coded to shielding. A similar weight has been transferred to structure cooling system.

TABLE IV

W-1 HEAT SHIELD

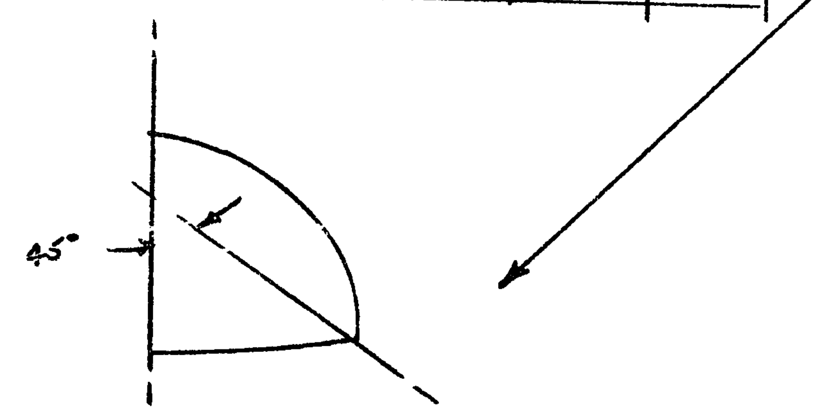
WEIGHTS (LB/FT^2) - NYLON PHENOLIC ABLATOR

(Local values shown, not to be used as averages)

$\gamma_c = 2.3$

	S/D = 106	W_{SA}	W_{INS}	W_{COOL}	W_{ABL}	TOT. W
VERTICAL	-1.4	1.0	.35	.35	.3	2.0
	-0.4	1.0	.35	.35	.3	2.0
	-0.2	-	.5	.5	2.8	3.8
	-0.1	-	.5	.5	8.0	9.0
	0	-	.5	.5	12.0	13.0
	+0.1	-	.5	.5	14.9	15.9
	0.2	1.0	.17	.17	12.3	13.6
	0.3	1.0	.17	.17	10.9	12.2
	0.4	1.0	.17	.17	7.8	9.1
	+0.6	1.0	.17	.17	3.4	4.7
	1.0	1.0	.17	.17	1.6	2.9
	+1.6	1.0	.23	.23	1.0	2.4
HORIZONTAL	0	-	.5	.5	12.0	13.0
	.1	-	.5	.5	10.4	11.4
	.2	-	.5	.5	7.1	8.1
	.3	-	.5	.5	2.0	3.0
	.4	1.0	.35	.35	.4	2.1
	.8	1.0	.35	.35	.3	2.0
	1.5	1.0	.35	.35	.3	2.0

	S/D = 106	W_{SA}	W_{INS}	W_{COOL}	W_{ABL}	TOT. W
POINT AROUND GIRTH AT 45° FROM VERT.	+0.4	1.0	.17	.17	13.8	15.1
	0.6	1.0	.17	.17	9.5	10.8
	1.0	1.0	.17	.17	4.5	5.8
	+1.6	1.0	.17	.17	2.8	4.1



TABLE

L-2C

WEIGHT BREAKDOWN

$r_{℄} = 2.3$

VERTICAL ℄						HORIZONTAL ℄					
S/R_o	W_{ABL}	W_{SA}	W_{INS}	W_{TCOOL}	W_{TOTAL}	S/R_o	W_{ABL}	W_{SA}	W_{INS}	W_{TCOOL}	W_{TOTAL}
≤-1.2	--	1.0	.23	.23	1.46	0	1.48	--	.50	.50	2.48
-1.0	.42	1.0	.27	.28	1.97	.2	1.43	--	.50	.50	2.43
-.8	.48	1.0	.26	.26	2.00	.4	1.38	--	.50	.50	2.38
-.6	.48	1.0	.26	.26	2.00	.6	1.33	--	.50	.50	2.33
-.4	.50	1.0	.25	.25	2.00	.7	1.52	--	.50	.50	2.52
-.2	.80	1.0	.25	.25	2.30	.8	2.24	--	.50	.50	3.24
0	1.48	-	.50	.50	2.48	.9	2.95	--	.50	.50	3.95
.2	1.54	-	.50	.50	2.54	1.0	2.24	--	.50	.50	3.24
.4	2.56	-	.50	.50	3.56	1.1	.47	1.0	.26	.26	1.99
.6	4.01	-	.50	.50	5.01	1.2	.24	1.0	.29	.29	1.82
.8	5.51	-	.50	.50	6.51	1.3	.13	1.0	.29	.29	1.69
.95	9.68	-	.50	.50	10.68	≥1.4	--	1.0	.23	.23	1.46
1.1	6.34	-	.50	.50	7.34						
1.2	4.08	-	.50	.50	5.08						
1.4	.15	1.0	.30	.30	1.75						
1.6	.15	1.0	.30	.30	1.75						
1.7	.15	1.0	.30	.30	1.75						
1.8	.13	1.0	.31	.31	1.75						
2.0	.13	1.0	.31	.31	1.75						
2.2	.11	1.0	.32	.32	1.75						
1.0	9.47	-	.50	.50	10.47						

2 Environmental Cooling Structure

The environmental cooling system includes a closed loop of ethylene glycol fluid, a radiator exposed to space and an open loop of water which is boiled off.

The closed loop is comprised of tubed sheet which forms the shell of the pressure structure and is coded to structure.

The cooling system is designed to keep the pressure structure at 200°F or less. An optimum cooling system size is theoretically supposed to weigh one-half of the total insulation required. The heat-shield weight includes the part of insulation weight allotted to shield weight.

Based on the above with a system factor of 2.3, a system weight of 126 lb for the W-1 and 117 lb for the L-2C would be indicated. However, a study made by the environmental group indicated a cooling system weight of 290 lb for a heat rate of ½ BTU/sec. This weight included 112 lb for boil-off water and had a system factor of 2.6.

A compromise was effected to arrive at a system weight by use of the following formula

$$\sqrt{\frac{W_{B_S}}{W_{B_C}}} \times W_{Sys.} + W_{B_S}$$

Where

W_{B_S} = Water Boil-Off Designed by Heat Shield Personnel

W_{B_C} = Water Boil-Off Designated by Environmental Personnel

$W_{Sys.}$ = Water System Less Boil-Off Water Developed by the Environmental Group.

	W-1	L-2C	W-1 Integ.	L-2C Integ.
Struct. Cooling Wt. ~ Lb	180	172	201	209

3 Structure

The structure has been designed as a pressure-type structure wherever possible. All curved structures form a part of a sphere except for the bottom bulkhead of the L-2C which is designed as a flat panel.

The pressure shell is of roll bond or of tubed sheet which holds the structure cooling water.

The structure required to hold the 12.2 PSI pressure is relatively light and near minimum gage in many areas. Frame spacing is 10" on W-1 and 18" on L-2C. There are 3 longerons on the W-1 and 4 on L-2C.

Critical problem areas of the W-1 and L-2C may be as follows:

<u>W-1</u> The intersection of the two spherical domes of the aft bulkhead may develop eccentric loading which may offset the benefit of a spherical shape. Some weight allowance has been included for this effect. The use of beams in this bulkhead is to take care of 3 PSI loads from external sources with no internal pressure.

<u>L-2C</u> The flat bottom bulkhead has been designed for strength. A rough check indicates deflection may be a problem and require larger beam areas than has been anticipated.

<u>W-1 and L-2C</u> The weight cost associated with making the APU a separate pressurized area has not been evaluated.

A detail weight breakdown of the four Apollo configurations is presented below:

	W-1	L-2C	W-1 Integ.	L-2C Integ
Command Module Struct.	(920)	(869)	(1156)	(1160)
Fwd Press. Bulkhead	-	52	54	
APU - Bulkhead	51	84		
Aft Press. Bulkhead (Bot)	232	451	293	
Side Press. Skins	255	62	293	
Frames	89	18	108	
Longerons	36	17	49	
Flooring	72	32	96	
Intercostals	25		28	
Door Frame and Mech.	25	29	25	
Seat Supports	30	30	30	
Console Supports	30	25	30	
Windows 3 at .33 ea.	15	10	15	
Separation Fittings	10		10	
Misc	50	59	75	
Camera Window	-		50	

4 Flaps

Flaps are located on the bottom and side of the Command vehicle and provide pitch and yaw control. The W-1 configuration with its split bottom flaps may be able to provide roll control.

The structural weight of the flaps has been derived through empirical formula. Heat shield requirements are based on 50% of stagnation temperature on the W-1 and 30% stagnation temperature on the L-2C.

	W-1	L-2C	W-1 Integ.	L-2C Integ.
Flap Structure	73	57		
Heat Shield	178	116		
Total Weight	251	173	320	211
Area Sq. Ft.	22	27	27	33

Derivation of Struct. Weight

W-1

$$^{w}b/_{s_b} = 2.31 \times 10^{-7}\ C\ (Los)^{2.4}$$

$$= 2.3 \times 10^{-7}\ (1.6)\ (800)^{2.4}$$

$$= 3.3\ lb/$$

L-2C

$$^{w}b/_{s_b} = 2.31 \times 10^{-7}\ C\ (Los)^{2.4}$$

$$= 2.31 \times 10^{-7}\ (2.5)\ (544)^{2.4}$$

$$= 2.11\ lb/$$

5 Surface Controls

Weight studies of a hot gas versus hydraulic power for the flap control system indicate the former method is lighter. This is primarily due to reliability factors. Whereas the hydraulic system has 100% redundancy, only the servovalves in the hot gas system require redundancy.

The hot gas system consists of a 400 PSI positive expulsion bladder-type tank. Tank pressure is supplied by a 4500 PSI helium source which also pressurizes the attitude control system. Operating temperature of the hot gas is 1700°F. Actuator pressure is assumed at 700 PSI. Another advantage of the hot gas system is that since it is located external to the pressurized compartment, it does not create heat within the module.

A weight breakdown of the hot gas systems is shown below.

The L-2C system is based on 3 surfaces, 140° deflection and a flap moment of 10,000 ft.lb. This system requires a radial actuator due to the larger flap movement.

The W-1 system is based on 4 surfaces, 42° deflection and a flap moment of 6640 ft.lb. This system has a lunar actuator.

The modular versions provide the ratios for the integrated versions by platform area of the Command Module.

Flap Control Weight Breakdown

	W-1	L-2C	W-1 Integ.	L-2C Integ.
Hot Gas System	(200)	(232)		
Harmonic Drives	-	84		
Vane Type Actuators	-	35		
Linear Actuators	60	-		
Servo Valves	8	6		
Transmission Lines	15	15		
Propellant (N_2H_4)	42	29		
Tanks	20	9		
Valves				
Relief	4	4		
Flow Control	4	3		
Gas Generator	5	5		
Filter	2	2		
Misc	5	5		
Pressure Plumbing & Valves	15	15		
Misc	20	20		
Controls	(58)	(58)		
Control Stick	6	6		
Rate Gyro Package	2	2		
Accelerometer Package	1	1		
Side Stick & Pedal Pick Offs	3	3		
Flight Control Units	43	43		
Wiring	5	5		
Total	(258)	(290)	(330)	(403)

6 Reaction Controls

The attitude control for the Command Module uses a hypergolic propellant of N_2O_4 and ½ UDMH/½ N_2H_4. Pitch control is obtained by 2 180 lb thrust rockets. Yaw and roll control is obtained by 4 90 lb thrust rockets. A complete backup system of lines and nozzles are carried for reliability. The propellant quantity of 80 lb (an assumed value from propulsion group) is carried in two ½ cubic foot spherical tanks. The tanks are positive expulsion types utilizing 4500 PSI helium gas as the expulsion force. The propellant tanks were estimated by using basic stress calculations for a tank pressure of 300 PSI. The helium bottle containing .6 pounds of helium gas is also used to pressurize the N_2H_4 tank of the flap control system. The weight of the 150 PSI chamber pressure nozzles, including injector head and actuators, was obtained from preliminary designs and weight estimates made by Rocketdyne.

	W-1	L-2C	W-1 Integ.	L-2C Integ.
Reaction Controls	160		205	225
Propellant	80			
Propellant Tanks	10			
Pressurization Gas and Tank	7			
Nozzles and Valves				
8-90 lb units	25	Same as W-1		
4-180 lb units	15			
Plumbing				
Propellant	14			
Pressurization	4			
Miscellaneous	5			

7 Landing System

The landing system is based on a threefold system, drogue chute, main chute and retro-rocket, although the drawings show landing bags rather than retro-rocket for the L-2C. The drogue chute is an 11 foot ribbon-type and is deployed at 85,000 foot altitude. The main, deployed at 10,000, is a 47 foot ringsail. Backup drogue and main chutes are carried for reliability. The final approach velocity of 70 ft/sec is absorbed by a single-stage solid propellant rocket utilizing two different thrust levels. The sizes and magnitudes of the components for the integrated versions have been ratioed from the modular L-2C and W-1. The weights of the various components have been estimated from vendor's data.

	W-1	L-2C	W-1 Integ.	L-2C Integ.
Landing System	450		575	630
Retro Rocket	210			
Mounting and Detacting Ring	18			
Cable 45' 5/8" 7x19 Cable with Shackles	32	Same as W-1		
2 Drogue Chutes with Risers	70			
2 Main Chutes	70			
Sequencing Equipment, Controls and Displays	50			

8 Auxiliary Power System

Electrical power is generated, when possible, by solar cells located on the Mission Module. During launch, guidance corrections, earth and moon shadow periods and re-entry, electrical power is generated by two 28 volts DC generators located in the Command Module. The generators, each of which is capable of generating the maximum average load of 2.2 kilowatts, are driven by oxygen hydrogen turbines. The turbines also drive coolant pumps required for re-entry. The mixture ratio is hydrogen enriched to keep the combustion temperature within limitation of the turbines. The electrical load requirements are:

Re-entry	3.65 KWH
Total Circumlunar Mission	22.04 KWH
Total Lunar Orbit Mission (14 Day Mission - 50 Mile Orbit)	46.50 KWH

The oxygen and hydrogen supplies are stored in a supercritical state in spherical or cylindrical tanks under 400 psia pressure. The conceptual arrangement of tanks for the modular designs is an oxygen tank in the Command Module large enough for the lunar orbit mission and off-loaded for the circumlunar mission. Because of space limitations a hydrogen tank sufficient for circumlunar mission only is located in the Command Module with a second tank added to the Mission Module when required for a lunar orbit mission. The second tank feeds into the primary tank as fuel is used. The drawings show tanks in the Command Module of insufficient volumes, but the weights are based on required volumes of 11.3 and 0.9 cu ft.

In the integrated versions the hydrogen tank in the Command Module is capable of lunar orbit and off-loaded for the circumlunar mission. A post landing cooling of 12 hours is obtained by a 1400 watt hour silver-zinc battery. A gasoline engine with fuel for 72 hour post-landing cooling has been discussed, but since the need is not clearly defined no weight is allowed for it.

The weight of the turbines with accessories and pump pad and the oxygen and hydrogen tanks were estimated from a similar design made by Airesearch during the DSI study. The other electrical components were estimated utilizing known weights of similar parts.

	W-1	L-2C	W-1 Integ.	L-2C Integ.
Auxiliary Power System	527	Same as W-1	675	Same as W-1 Integ.
Electrical Generators and Regulators	56		56	
Protective Relays	18		24	
Ext. Power Receptacle	2		2	
Circuit Breakers	6		6	
Bus Distribution Panels	18		33	
Recovery Battery	40		40	
Battery			18	
Voltage Boosters	10		15	
Voltage Regulator			6	
Terminals, Connector	20		30	
Wiring	20		40	
Installation	30		38	
H_2 Tank 2.89 Ft Dia	72		114	
H_2 Unusable	18		36	
Usable	32		32	
O_2 Tank 1.26 Ft Dia	19		19	
O_2 Unusable	11		11	
Usable	24		24	
APU Units w/Accessories and Pump Pad	106		106	
Plumbing	15		15	
Exhaust	10		10	
Additional Propellant Required for Lunar Orbit	24		56	

9 Environmental Controls

The atmosphere of the living compartments is a mixture of nitrogen and oxygen. The total pressure is equal to a 5,000 altitude (12.2 PSIA) with the oxygen partial pressure equal to that of sea level condition. Oxygen is replenished at the same rate that CO_2 is abosrbed from the cabin. The maximum design CO_2 concentration is 13 times that of sea level condition. Oxygen and nitrogen is provided to make up loss of atmosphere for the following assumed conditions.

(1) A leak rate of 0.05 pounds per hour.

(2) Two recharges from a 5.5 PSIA to the desired pressure of 12.2. The loss of pressure is assumed to be due to meteorite penetration and the leak would be repaired by the time pressure had dropped to 5.5 PSIA.

Oxygen and nitrogen is stored in 6 separate containers with 4 of them being cryogenic storage. In the modular designs the large oxygen and nitrogen containers are located in the Mission Module. Reliability is obtained during re-entry by having both oxygen and nitrogen stored in the containers. The weights of the oxygen and nitrogen tanks were estimated by comparing them with known or preliminary designs.

CO_2 is removed, while in free flight, by rechargable molecular sieves and during re-entry by lithium hydroxide. Two molecular sieves are used so that one can be in use while the other is being recharged. In case of failure of one unit, CO_2 concentration will be allowed to increase during the recharging cycle. The weight of the molecular sieves is based on vendors' estimates.

Water is removed from the cabin's atmosphere by absorbent sponges and with the urine water is reclaimed for crew's drinking water. Two recovery units are carried in the Mission Module for reliability. The weight of these units is based on vendors' preliminary data.

Cooling of crew and equipment is obtained by 130 sq. ft. of space radiator located on the Mission Module. During launch, re-entry and other periods of radiator inoperatives, cooling is obtained by evaporative cooling in a water boiler. Cooling during prelaunch is obtained from a ground based system utilizing freon. The weight of the space radiators is based on 0.040 roll bond aluminum tube sheet. The weight of the radiators is 1 psf which includes fluid in tubes and attachment.

Post landing cooling is by circulating fans for 24 hours. Power is from the recovery battery included in auxiliary power system. Although post landing cooling of 72 hours has been talked about, no weight has been allotted for this purpose.

	W-1	L-2C	W-1 Integ.	L-2C Integ.
Environmental Control	310		973	
Oxygen System				
12" D. Sphere	3		3	
6" D. Sphere	8		8	
17" D. Sphere			42	
Oxygen	19		128	
Nitrogen System				
12" D. Sphere	3		3	
12" D. Sphere	13		13	
15" D. Sphere			25	Same as W-1 Integrated
Nitrogen	32	Same as W-1	83	
Oxygen & Nitrogen Plumbing	10		30	
Initial Charge of Air	23		54	
Filter Assembly Li H	27		27	
Molecular Sieve			61	
Cooling System Components				
Heat Exchanger	15		30	
Water Separator	3		6	
Fans	6		11	
Controls	4		9	
Ducts	12		25	
Space Radiator Lines & Fluid	20		110	
Water Recovery Unit			150	
Re-entry and Survival System				
Water	52		52	
Water Stowage & Explusion	16		16	
Snorkel	14		14	
Lines, Fittings, Etc.	20		40	
Supports	10		33	

10 Instruments

The instrument weights were estimated partially by the guidance group and partially by the weights section. The items attitude through position computer listed below were estimated by the guidance group. This includes the structural panel and controls located on the panel. Therefore, the panels and consoles listed under funishings were decreased, and the weight for controls of the various systems were deleted or decreased. The weight of instruments will be higher than what would be expected. When more details are known the weight of these panels can be separated into their respective categories.

The balance of the instruments was estimated by using known or similar instruments. The weight of these items are only for the display portion of the panels.

	W-1	L-2C	W-1 Integ.	L-2C Integ.
Instruments	260	260	260	260
Attitude (2)	16	16	16	16
Thrust	5	5	5	5
Time	3	3	3	3
Ranging System	5	5	5	5
Guidance Evaluation (2)	28	28	28	28
Position Indicator (2)	16	16	16	16
Velocity Correction (2)	16	16	16	16
Subsystem Status (2)	16	16	16	16
Data Retrieval	5	5	5	5
Vehicle Measurement	10	10	10	10
Reaction Panel (2)	10	10	10	10
Radio Guidance & Data Entry (2)	10	10	10	10
Position Computer	2	2	2	2
Periscope	60	60	60	60
Auxiliary Power	1	1	1	1
Electrical System	2	2	2	2
Hot Gas System	1	1	1	1
Environmental Control	15	15	15	15
Module Separation	2	2	2	2
Recovery	2	2	2	2
Launch Escape	2	2	2	2
Suit Pressure (3)	5	5	5	5
Alarm (4)	2	2	2	2
Solar Flare Warning (2)	2	2	2	2
Recording Equipment	2	2	2	2
Installation, Wiring	22	22	22	22

11. INSTRUMENTATION

The instrumentation system consists of components to collect and record data and, if required, to condition these signals for transmission. The transmitters are listed under communication and tracking. The number of end instruments has been assumed to be 50 with 30 of them located in the Command Module in the modular designs.

	W-1	L-2C	W-1 Integrated	L-2C Integrated
Instrumentation	98	Same as W-1	122	Same as W-1 Integrated
Tape Recorder	9		18	
Instrumentation Package	6		6	
Temp Measuring System	3		3	
Camera 35 mm.	3		3	
PCM System	20		20	
Data Link System	20		20	
Signal Conditioning Package			5	
Power Supply	1		1	
End Instruments	6		10	
Wiring, Support	30		36	

12. COMMUNICATION AND TRACKING

For the modular designs, the communication, tracking and telemetering equipment for the Command Module is comprised of the following units with their designated uses:

(1) VHF transmitter (2). Used for range T/M and communication.

(2) VHF receiver (2). Used for range T/M and communication.

(3) UHF transmitter. Used for rescue communication.

(4) UHF receiver. Used for rescue communication.

(5) HF transmitter. Used as a rescue beacon.

(6) SHF transmitter. Used for communication during re-entry.

(7) C-Band tracking beacon. Used for tracking during boost, pre and post re-entry.

Items 1 and 2 have two units each for reliability. Other tracking aids are included in this group.

The weights of the components were estimated by the communication group and are based on like systems. The sofar bombs were assumed from Mercury's known weight.

	W-1	L-2C	W-1 Integrated	L-2C Integrated
Communication and tracking	134	Same as W-1	179	Same as W-1 Integrated
UHF Transmitter	3.0		3.0	
UHF Receiver	.5		.5	
HF Transmitter	2.0		2.0	
Recovery Antenna Package	15.0		15.0	
Sofar Bomb (2)	4.0		4.0	
Dye Marker	1.0		1.0	
Sea Water Battery	4.0		4.0	
Flashing Lights	7.0		7.0	
SHF Transmitter	30.0		30.0	
C-Band Transmitter	4.0		8.0	
Radar CHFF	1.5		1.5	
Servo Amplifier	1.0		1.0	
VHF Transmitter (2)	2.0		2.0	
VHF Receiver (2)	1.0		1.0	
VHF Flush Antenna (4)	20.0		20.0	
SHF/C Band Antenna (2)	6.0		8.0	
Control Panel	1.5		1.5	
Antenna Multiplexer	.5		.5	
S Band Transmitter (2)			6.0	
S Band Receiver (2)			1.0	
VHF Transponder			1.0	
X Band Beacon			5.0	
Wiring Support	28.0		36.0	

13. GUIDANCE

All of the guidance components are located in the Command Module. The weights of the various components were estimated from known systems under development or from similar equipment.

Primary guidance is obtained by an astro-inertial platform combined with an optical tracker. A miniature platform with an auto-manual tracker is used as a backup. Each system has its own digital computer. A radio altimeter is used for altitude determination during orbital phases and at re-entry. The weight of the altimeter was taken from DS-1.

Radio guidance is obtained by utilizing the VHF Transponder included under communication and tracking.

	W-1	L-2C	W-1 Integrated	L-2C Integrated
Guidance	220	Same as W-1	Same as W-1	Same as W-1 Integrated
Astro-Inertial Platform with Tracker & Electronics	55			
Miniature Platform	13			
Miniature Platform Electronics	12			
Digital Computer (2)	40			
Automanual Tracker	10			
Automanual Tracker Electronics	10			
Radio Altimeter	40			
Digital Data Decoder	1			
Voice System Detector	1			
Receiver System	3			
Installation	35			

14 SCIENTIFIC EQUIPMENT

The scientific equipment carried is used mainly for aero-medical and visual observation of the moon's surface. The weight of the various components was estimated by the instrumentation group. For the modular version, most of the equipment is located in the Mission Module. **This latest estimate exceeds the weight allowance for scientific equipment by 100 lb.**

	W-1	L-2C	W-1 Integrated	L-2C Integrated
Scientific Equipment	70	Same as W-1	175	Same as W-1 Integrated
Camera			100	
Film	30		30	
Radiation Detector (4)			12	
Micrometeorite Detector (3)			15	
Solar Flare Sensor (2)			8	
RBE Measurement	6		6	
TV And Electronics	8		8	
Experiment Stowage			30	
Control Panels	10		10	
Wiring, Misc.	16		55	
Difference allowance to est.			**-100**	

15. FURNISHING AND EQUIPMENT

The furnishings and equipment carried in the Command Module for the modular versions is that which is required for crew functions. One-day supply of food and water is stowed in the Command Module with the balance of food stowed in the Mission Module.

	W-1	L-2C	W-1 Integrated	L-2C Integrated
Furnishing and Equipment	239	Same as W-1	413	Same as W-1 Integrated
Seat and Restraints (3)	75		75	
Panels and Consoles	40		40	
Bunk			30	
Survival Kit	60		60	
Food	4		58	
Water	30		30	
Containers	3		8	
Waste Disposal	2		22	
Spare Parts and Tools	25		25	
Flooring			40	
Support			25	

16. CREW

The crew weight is based upon a 75 percentile man of 176.6 lb. plus 33.4 lb. for personal gear and space suit.

3 at 210 lb. each 630 lb.

C. MISSION MODULE

1. Structure

The Mission Module structure is identical for all modular designs except for the fairing between the Command and Mission Modules. The weight actually includes all structure aft of the Command Module, including the Mission Module pressurized capsule.

The Mission Module structure for the integrated versions are similar in design except they do not contain the Mission Module pressurized capsule and consequently are shorter in length.

The outer shell of the 154" dia. cylindral shape consists of .040 aluminum skin with 12 stringers of 0.36 □" crosssection, 12 int. stringers of 0.14 □" and 0.40 GA frames at 30" spacing. An .040 titanium bulkhead at the aft end acts as a flame shield and side support for the main engine.

The pressurized crew capsule is a cylinder with two dome ends. The skin is .040 gage, there are .040 frames at 18" spacing along the cylinder portion and 12 stringer of 0.05 □". There is a window in one dome and a door in the other dome. The capsule is covered with ¼" LINDE super I insulation.

A weight penalty of 98 lb. is included in all Mission Modules. This weight increase is due to extending the length of the Mission Module skirt to take care of the propellant tanks sized for lunar takeoff.

Mission Module Structure Breakdown

	W-1	L-2C	W-1 Integ.	L-2C Integ.
Outer shell	(765)		(671)	(598)
Skin	250		190	151
Stringers	102		102	72
Frames	70		69	80
Flame shield	152		152	152
Conical Tank Support	69		--	--
Misc.	24		60	45
Δ For Incr. Length	98		98	98
Pressure Module	(329)		(--)	(--)
Skin	171			
Stringers	5			
Frames	14			
Window (1)	15			
Door (1)	30			
Flooring	37			
Misc.	24			
Insulation	33			
Total - Common Parts	(1094)			
Attach. to Command Module	(167)	(110)		
Skin	98	59		
Frames	32	30		
Tunnel	37	11		
Misc.	--	10		
Total Mission Module	(1261)	(1204)	(671)	(598)

2. Reaction Controls

A hypergolic propellant mixture of N_2O_4 and 1/2 UDMH/½N_2H_4 is contained in the Mission Module to control the altitude of the spacecraft during free flight and moon orbit, and to supply power for guidance corrections. The altitude control nozzles consist of two systems with each system having two 30 lb. pitch and four 15 lb. yaw and roll nozzles. The vernier nozzles are two 300 lb. thrust units. They are canted 20 degrees so as to minimize pitching due to one engine failing to start.

The propellants are stored in two 6.5 cu. ft. spheres. The tanks are pressurized to 300 psi by a 4500 psi helium source. The present size of the tanks are not sufficient for the required propellant. It will be necessary to enlarge the tanks to approximately 7 cu. ft. each. Nozzles were estimated from study made by Rocketdyne.

Propellant required for altitude control is based on the number of times the spacecraft will be reorientated for solar cell alignment, star tracking and guidance correction. The total number of turns was calculated to be 201. For the assumed turning rate of 10 deg/sec., this required 241 lb. of propellant. Including propellant for limit cycle and corrective impulse, the total amounts to 279 lb. This was arbitrarily decreased by 35 lb. when a radiation shield was added.

The vernier fuel was calculated for velocity increments required for guidance corrections.

For all the missions, propellant weight equivalent to three percent of maximum volume was considered trapped in the tanks and lines.

	W-1	L-2C	W-1 Integrated	L-2C Integrated
Reaction Control	1 044			
Nozzles				
8 Roll & Yaw 15# T.	10			
4 Pitch 30# T.	6			
2 Vernier 300# T.	30			
Tanks - 2 at 6.5 cu. ft.	76	Same as W-1	Same as W-1	Same as W-1
Pressurization	43			
Plumbing and Controls	25			
Propellant				
Trapped	27			
Ullage	7			
Altitude Control	244			
Vernier - Circumlunar	576			

3. Auxiliary Power System

Electrical power is primarily derived from 288 sq. ft. of solar cells. These cells are arranged in 12 panels folded into the surface of the Mission Module during launch and, once in space, are erected by rotating them 90 degrees about their aft end. During launch the panels are held down by their hinges on the aft end and a circumferential band located 30 % from the free end. The band is segmented into three sections of 120° each and are connected together by explosive bolts. The solar cell arrays were calculated from stress requirements for the skins and core. The skins vary from 0.008 to 0.016 aluminum and the core is 3/8 x 0.001 and 3/8 x 0.002 aluminum honeycomb. The structure including hinges, springs, and tiedown strap weighs 0.969 p.s.f. The cells including bond and wiring weighs 0.51 p.s.f.

In the Modular concept, the Mission Module contains a booster battery and power distribution for the equipment located in this module. For the lunar orbit mission, an additional hydrogen tank is installed in the Mission Module.

In the integrated version all of the components with the exception of the solar cells are moved to the Command Module.

	W-1	L-2C	W-1 Integrated	L-2C Integrated
Auxiliary Power System	514	Same as W-1	426	Same as W-1 Integrated
Battery	18			
Voltage Booster	5			
Voltage Regulator	6			
Relays	6			
Distribution Panel	15			
Wiring	30			
Installation	8			
Solar Cell Array	411		411	
Solar Cell Tie Down	15		15	
Additional Auxiliary Power Fuel Required for Lunar Orbit	127			
H_2 Tank and Plumbing 2.89' Dia.	77			
H_2 Unusable	18			
Usable	32			

4. Environmental Controls

The environmental control system has been briefly described under Command Module weight derivation. Due to space limitations the large components are located in the Mission Module. The integrated version of Command Module contains all components of this system with the exception of the space radiators. It was assumed that the same components would be required for the integrated versions as for the modular ones. Some weight would be saved in the integrated version since duplication would not be required to the extent that it is in the modular concept.

	W-1	L-2C	W-1 Integrated	L-2C Integrated
Environmental Control	798	Same as W-1	135	Same as W-1 Integrated
Oxygen System				
17" D. Sphere	42			
Oxygen	109			
Nitrogen System				
15" D. Sphere	25			
Nitrogen	51			
Oxygen & Nitrogen Plumbing	20			
Molecular Sieve	61			
Cooling System Components				
Heat Exchanger	15			
Water Separators	3			
Fans	5			
Controls	5			
Ducts	13			
Water Recovery Unit	150			
Space Radiators	135		135	
Space Radiators Lines & Fluid	90			
Lines, Fitting, etc.	20			
Supports	23			
Initial Charge of Air	31			

5. <u>Instrumentation</u>

The only instrumentation components located in the Mission Module are a tape recorder, signal conditioning package, and twenty end instruments.

	W-1	L-2C	W-1 Integrated	L-2C Integrated
Instrumentation	24		0	0
Signal Conditioning Package	5			
End Instruments	4	Same as W-1		
Tape Recorder	9			
Wiring, Support	6			

6. Communication and Tracking

In the modular designs, communication, tracking and telemetering equipment for launch and deep space probes are contained in the Mission Module. In the integrated version all of these, with the exception of the antenna, have been moved to the Command Module. The equipment required for launch and deep space probes are listed below with their associated functions.

(1) S-Band transmitter (2). Used for DSIF telemetering and communication.

(2) S-Band receiver (2). Used for DSIF telemetering and communication.

(3) VHF transponder. Minitrack response and communication.

(4) C-Band transmitter. Used for tracking during boost.

(5) X-Band Beacon. Used for tracking during boost.

	W-1	L-2C	W-1 Integrated	L-2C Integrated
Communication and Tracking	55.0	Same as W-1	30.0	Same as W-1 Integrated
S-Band Transmitter (2)	6.0			
S-Band Receiver (2)	1.0			
S-Band Monopole Antenna (4)	4.0		4.0	
S-Band Parabolic Antenna (2)	16.0		16.0	
VHF Transponder	1.0			
C-Band Transmitter	4.0			
X-Band Transmitter	5.0			
C/X Band Antenna (4)	8.0		8.0	
Wiring, Support	10.0		2.0	

7. Scientific Equipment

Most of the scientific equipment for the modular versions is located in the Mission Module. In the integrated versions it was all considered to be located in the Command Module, but the various sensors will probably have to remain in the propulsion module. This latest estimate exceeds the weight allowance for scientific equipment by 100 lb.

	W-1	L-2C	W-1 Integrated	L-2C Integrated
Scientific Equipment	105	Same as W-1	0	0
Camera	100			
Radiation Detector (4)	12			
Micrometeorite Detector (3)	15			
Solar Flare Sensor (2)	8			
Experiment Storage	30			
Wiring, Misc.	40			
Less				
Differance allowance to est.	-100			

8. Furnishing and Equipment

Food for three men for thirteen days is stored in the Mission Module. It is assumed to be the multi-metal tube (MMT) diet of 1.38 lb. per/man per/day.

	W-1	L-2C	W-1 Integrated	L-2C Integrated
Furnishing and Equipment	174	Same as W-1	0	0
Food	54			
Containers	5			
Waste Disposal	20			
Bunk	30			
Flooring				
26.6 sq.ft. X 1.5 #/sq.ft.	40			
Equipment Supports	25			

D. PROPULSION

The main propulsion system is used for major guidance corrections, injection and ejection from a lunar orbit, turn around and lunar takeoff. The tanks have been sized for the lunar takeoff which is the mission that requires the largest velocity increment.

The tankage as shown on the inboard drawings is made up of a 395 cu. ft. toroidal hydrogen tank and a 120 cu. ft. conical oxygen tank. For the present weight estimate, it will be necessary to increase the capacity of the tanks by five percent. The hydrogen tank was estimated assuming an average thickness of 0.072 aluminum and one in. of Linde Super I insulation at 4.7 lb./cu. ft. Later analysis indicates the insulation should be 0.5 in. of Linde Super I and 0.3 in. of styrofoam. The oxygen tank was estimated using gauges as supplied by stress. The oxygen tank is covered by 0.5 of insulation.

The main propulsion unit is the P. and W. XLR 119 with an extra inducer so that the engine will function with a NPSH of zero at the tanks. The assumed Isp of 430 was used for propellant calculations. The main engine is gimbaled two degrees. A weight allowance of 40 lb. is carried for this purpose. This allowance is based on first stage Vanguard gimbaling controls which are similar to Apollo's requirements.

The estimated 100 lb. of plumbing is based on the plumbing system of Saturn C-2 SV stage.

The unusable propellant is based on engine manufacturer specifications and previous design experience. The amount of unusable propellant varied for different missions and was estimated using the following assumptions.

Type of Loss	Amount of Loss H_2	Amount of Loss O_2
Trapped in tank and retained vapor	2% Tank Capacity	0.4% Tank Capacity
Trapped in Engine	1 lb.	5 lb.
Trapped in Lines	2 lb.	30 lb.
Startup / Engine Start		
Precooling	5 lb.	10 lb.
Startup	7 lb.	35 lb.
Decay	1 lb.	5 lb.
Boiloff		
Circumlunar	3.5% Tank Capacity	0.5% Tank Capacity
Lunar Orbit	7% Tank Capacity	1% Tank Capacity
Lunar Takeoff	3.5% Tank Capacity	0.5% Tank Capacity
Outage	.5% Usable Amount	

For the W-1 design this amounts to the following unusable propellant for the three missions:

	Circumlunar	Lunar Orbit	Lunar Takeoff
Trapped in tank and retained vapor.	70	70	70
Trapped in engine	6	6	6
Trapped in lines	32	32	32
Startup	63	189	126
Boiloff	104	208	104
Outage	3	6	10
Total unusable	278	511	348

	W-1	L-2C	W-1 Integrated	L-2C Integrated
Propulsion	(1348)			
Engine	330			
Hydrogen Tank				
Skin, frame, etc. 367 sq ft x 1.04 lb/ sq ft	382			
Insulation 367 sq ft x .39 lb/sq ft	143			
Baffle	5			
Oxygen tank				
Dome 9220 x .051 x 0.1	49	Same as W-1	Same as W-1	Same as W-1
Cone 9504 x .080 x 0.1	78			
Angle	10			
Tank fttg	10			
Baffle	15			
Misc. Stiffeners	15			
Insulation 130 sq ft x .2 lb/sq ft	26			
Gimbaling Control	40			
Tank and Engine Support	145			
Plumbing	100			

W-1

Summary of Propellants

	Circumlunar	Lunar Orbit	Lunar Takeoff
Command Module			
Flap Control N_2H_4	42	42	42
Auxiliary Power	(85)	(109)	(109)
H_2 Unusable	18	18	18
Usable	32	32	32
O_2 Unusable	11	11	11
Usable	24	48	48
Reaction Control	(80)	(80)	(80)
N_2O_4	53	53	53
UDMH / N_2H_4	27	27	27
Mission Module			
Auxiliary Power		(50)	(50)
H_2 Unusable		18	18
Usable		32	32
Reaction Control & Vernier	(854)	(1123)	(388)
Trapped	27	27	27
Ullage			
N_2O_4	5	13	5
UDMH / N_2H_4	2	7	2
Attitude Control			
N_2O_4	163	163	82
UDMH / N_2H_4	81	81	40
Verniers			
N_2O_4	384	555	155
UDMH / N_2H_4	192	277	77
Main Propulsion	(2984)	(6588)	(10672)
H_2 Unusable	115	205	135
Usable	451	1013	1721
O_2 Unusable	163	306	213
Usable	2255	5064	8603

Design Criteria

TM-23

[redacted]

Issue No. 2
3-13-61

Preliminary

Design Specification

for Apollo Spacecraft

Technical Memorandum 23

R. R. Drummond

13 March 1961

[redacted]

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[illegible]

9.0 Ground Support Equipment

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10.2 Flight Test Program

1.0 General Spacecraft Requirements:

1.1 Mission Objectives:

1.11 The basic mission objectives are to prove the feasibility of practical manned space flight by successfully developing a spacecraft with the following capabilities.

1.111 To navigate earth-lunar space with a crew of 3 men.

1.112 To perform safely a planned space mission from earth launching.

1.113 To return and re-enter the atmosphere safely up to maximum velocity.

1.114 To execute a safe landing in the area of a predetermined site on either land or water.

1.115 To provide for a period of crew survival after landing of not less than 72 hours.

1.12 Basic mission flight plans are evolved from the following:

1.121 Lunar reconnaissance flights

1.1211 Cislunar or circumlunar flight

1.1212 Lunar orbit

1.1213 Lunar Landing (Projected future, but not a present requirement)

1.122 Earth orbit flights.

1.1221 Crew training in orbit

1.1222 Scientific experiment

1.1223 Orbit rendezvous (Projected future, but not a present requirement)

1.13 Duration of Flight

1.131 The spacecraft shall be capable of a maximum duration of mission of 14 days.

1.132 In event of an emergency such as a solar flare immediately following injection into lunar trajectory, the maximum duration of space flight under radiation exposure shall be 5 days, assuming full utilization of the abort capability incorporated into the spacecraft.

1.133 Nominal circumlunar mission flight time shall be 170 $\pm$ 10 hours.

1.2 Mission Abort Capability

In event of malfunction of systems, equipment, or instruments, or for any other emergency requiring an abort of the mission, or escape by the crew, the following capabilities are incorporated into the spacecraft.

1.21 On Launch Pad or During Boosted Flight: Crew escape prior to or during boosted flight by escape and/or separation rockets.

1.22 Outgoing Leg of Lunar Trajectory: Retro-capability to arrest flight duration and distance after injection in the outgoing leg of a lunar trajectory.

1.23 Lunar Orbit: No abort capability except normal ejection from lunar orbit on the next circuit around the moon.

1.24 Return Leg of Lunar Trajectory: No abort capability except midcourse guidance corrections on a return leg from a lunar trajectory.

1.25 Re-entry: Re-entry abort capability by means of a skip out maneuver in case of positioning or re-entry angle errors. This allows a second chance at re-entry positioning.

1.3 Launch Boosters:

1.31 Earth Orbit Flights: Saturn C-1 Booster

1.32 Lunar Trajectories:

Saturn C-2

1.33 Booster Payload Design Limitations:

Saturn C-1 (Earth orbit)
Saturn C-2 (Escape)

2.0 General Design Concepts

2.1 Design Objectives:

2.11 The spacecraft and subsystems shall be designed for the basic lunar orbit mission with consideration for future growth to the lunar landing mission. Propellant tank volume and module size for lunar take-off shall be intially incorporated.

2.12 For earth orbit training flights and the circum-lunar or lunar orbit missions, the basic spacecraft shall be in an offloaded condition consistant with booster payload limitations.

2.2 Internal Environment:

2.21 A shirt sleeve environment shall be provided for the crew in space flight.

2.22 Space suits are also provided for the crew for use during launch, re-entry, or during emergencies.

2.23 A cabin altitude of about one mile (12.2 psi) shall be maintained as the minimum air pressure required to maintain maximum sustained working efficiency for the crew.

2.3 Module Concept

2.31 The spacecraft shall incorporate specific modules that may be expended or disposed of as the mission progresses, similar to staging of booster rockets.

2.31 The subdivision into modules shall be done in the most expedient manner consistant with maximum efficiency for performing the overall mission objectives.

2.33 Currently, the module breakdowns being considered are as follows:

Escape
Propulsion
Lunar Orbit Mission
Command Center (also Re-entry Module)
Lunar Landing (Future growth)

PRIORITY DEFINITIONS

2.51 First priorities are assigned to those functions necessary for meeting the mission primary objectives. Any equipment, systems, or instrumentation whose malfunction would jeopardize attainment of these objectives would fall in this category. If malfunction or failure occurred prior to launch, this would be cause to hold or scrub the flight, until a fix is made. If malfunction occurs in flight, and a quick fix cannot be made, this would be cause to abort the mission. First priority items will be designed for the highest probability values of successful operation.

2.52 Second priorities are assigned to those functions where redundancy exists or where non-vital parameters are measured or controlled, such as measurement of overall performance of the spacecraft or components thereof. Equipment, systems, or instrumentation whose malfunction would jeopardize only superficially, the attainment of the full mission objectives will be assigned second priority. If one of these malfunctions occurs prior to final fueling of the booster, it shall constitute cause to hold or scrub the flight. If the malfunction occurs after final fueling of the booster, a hold shall be at the discretion of the commanding officer of the spacecraft or the NASA Operations Director.

2.53 Third priority is assigned to functions associated with gathering data for future flights or for supplementary data for overall flight evaluation. Measurements other than the minimum required for a successful mission would fall in this category. Malfunctions of equipment, systems, or instruments in this case would only affect extra data gathering capability over and above the mission requirements. Third priority malfunctions will not require a hold after countdown has begun. Fixes on a non-interference basis may be made at the discretion of the NASA Operations Director.

2.7 Design Weights

2.71 Launch Weight - Total weight of the spacecraft at time zero.

2.711 Maximum Allowable Weights

2.72 Launch Escape Weight Prior to Injection: Launch weight of command module plus launch escape rocket and support. Mission module and space propulsion systems are not included.

2.721 Maximum allowable weight =

2.73 Mission Abort or Turn Around Weight - Total weight of spacecraft less launch escape rocket and support.

2.731 Maximum allowable weight =

2.74 Lunar Injection Weight: Total weight of spacecraft less the following:

Launch escape rocket and support
Vernier engine propellant for ΔV = 225 ft/sec.

Food, water, environmental control items, or other expendables that may be expelled from the spacecraft in the first 72 hours following time zero.

2.741 Maximum allowable weight =

2.75 Lunar Ejection Weight: Lunar injection weight less:

Space propulsion propellant for ΔV = 2750 ft/sec.
Vernier propellant for = 100 ft/sec.

2.751 Maximum allowable weight =

2.76 Re-entry Weight: (At beginning of re-entry for both abort and normal re-entry)

Launch weight of command module alone

2.761 Maximum allowable weight =

2.77 Landing Weight, Drogue and Main Chute Deployment:

2.771 Escape Condition: Launch weight of command module only.

2.772 Maximum allowable weight =

2.773 Normal Re-entry: Launch weight of command module less weight of expended ablation material and coolent.

2.774 Maximum allowable weight =

2.78 Landing Weight, Earth Impact

2.781 Escape Condition: Launch weight of command module less drogue and main chute weight.

2.782 Maximum allowable weight =

2.783 Normal Re-entry: Launch weight of command module, less:

Drogue and main chute weights, expended ablation material, and coolent weight.

2.784 Maximum allowable weight =

2.79 Flotation Weight, Water Landing

Landing weight-earth impact, less detachable portions of residual heat shield.

2.791 Maximum allowable weight =

3.1 Service Life and Malfunction Criteria

The design of the spacecraft shall be based on a philosophy of maximum reliability to complete the mission or to return safely in event of a malfunction. This is done first by setting up realistic design service life goals for components and systems and then by providing redundancy sufficient to isolate and constrain single failures to prevent growth into catastropic failure.

3.11 SERVICE LIFE DESIGN GOALS

3.111 Structure:

The basic structure shall be operated for the following number of missions:

Expendable structure	1 mission
Recoverable structure	5 missions

The structural design life span shall be determined as follows.

Design Life = $n_M K_T K_L$

n_M = number of missions

*K_T = factor for equivalent accumulated test time = 1.25

**K_L = Factor for life prediction = 1.5

Design Life = Expendable = 1 X 1.25 X 1.5 = 1.88 missions
Recoverable = 5 X 1.25 X 1.5 = 9.37 missions

3.112 Functional Systems:

In order to establish the design life goals for functional systems the following assumptions are made:

*K_T = Factor for accumulated equivalent test time

K_T = 1.5 for continuous operating systems

K_T = 2.0 for intermittent operating systems.

**K_L = Factor for life prediction = 1.5

C_T = Ratio of "on" time to total mission time

t = Total time for maximum mission, (hrs.)

n = Number of design missions

Design life = $nC_T t K_T K_L$ (hrs)

* K_T represents approximate expended environmental life prior to launch.

** Tolerance on ability to predict fatigue life of a composite structure is approximately ±50% based on aircraft criteria.

3.113 DESIGN LIFE SPANS FOR FUNCTIONAL SYSTEMS

Type System	Component Systems	n	C_T	t	K_T	K_L	Design Life (hrs)
Continuous Operating	Expendable	1	1	336	1.5	1.5	760
	Recoverable	5	1	336	1.5	1.5	3800
Intermittent Operating	Expendable	1	Varies with each system	336	2.0	1.5	1000 C_T
	Recoverable	5	Varies with each system	336	2.0	1.5	5000 C_T

3.12 Malfunction Criteria for Design

Wherever practical and feasible, the design philosophy for the spacecraft shall be as follows:

Single malfunctions shall be considered in determining critical design conditions.

The spacecraft shall be designed such that sequential type failures are prevented to as great a degree as possible. This shall be approached by providing sufficient redundancy in systems and structure to isolate and confine failures.

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3.2 Load Factors:

3.21 The following limit load factors shall be utilized for design of the spacecraft.

LOAD FACTOR ITEM		MAXIMUM LIMIT g's.
Boost Phase:		
Axial - Saturn C-1, first stage burnout		6.5
Axial - Saturn C-2, second stage burnout		5.2
Escape:		
Axial - Escape Rocket from launch pad (Command module & escape tower only)		20
Space Flight:		
Lunar injection Wt. = 15,000#	Engine 17,500# Thrust	1.2
Lunar ejection Wt. = 13,500#	Engine 17,500# Thrust	1.3
Mission abort Wt. = 13,500#	Engine 17,500# Thrust	1.3
Return leg guidance Wt. = 13,000	Engine 17,500# Thrust	1.35
Reentry:		
Resultant airloads, command module only (Maximum undershoot angle)		10
Chute Deployment:		
Drogue chute		5
Main chute		3
Landing:		
On land	Command module with landing bags	16
On water	Command module with landing bags	24
Ground Handling and Hoisting:		
Components		2
Total spacecraft		2.67
Equipment Containment:		
Cabin area crew protection		67

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3.22 Transportation Load Factors

The following load factors shall be applied to the shipping container supports with the spacecraft packaged for transport.

3.221

3.23 Equipment Mounting Load Factors - Sonic Vibration Criteria.

Presented herein is a preliminary estimate of equipment support load factors based on a random sonic vibration input derived basically from Titan I flight vibration data and adapted to the Apollo vehicle. Consideration is given to the following noise sources.

S-1 Saturn booster noise field. Aerodynamic noise at q_{max} upon ascent. Aerodynamic noise at q_{max} upon re-entry. Sea level escape rocket noise field.

3.231 Spacecraft Sonic Vibration and Limit Load Factor Data

Condition	Sound Pressure Level - db		Vibration Level, g rms		Equipment Support Structure Load Factor, g	
	External	Internal	Mission Module	Command Module	Mission Module	Command Module
Launch	145	125	15	5	40	15
Max q - ascent	144	124	20	5	50	15
Max q - re-entry	145	125	-	5	-	15
Sea level escape	170	150	-	35	-	90
Space flight (Engine induced vibration)			5	5	15	15
Space flight (Engine mounted structure)			45	-	110	-

These load factors decrease linearly with increasing equipment weight for weights in excess of 50 lbs. These factors apply only to acoustically and mechanically induced vibrations and do not include acceleration and transient shock loads.

3.31 PRESSURIZATION

Interior pressurization of the inhabited volume of the spacecraft shall be maintained as follows:

3.311 Upon entry of the crew into the spacecraft for final countdown, the vehicle shall be pressurized to 12.2 psi above sea level ambient.

3.312 A constant differential pressure of 12.2 psi shall be maintained across the vehicle pressure wall during exit and space flight. Upon re-entry the differential pressure system is deactivated except for pressure relief valves.

3.313 Upon deployment of the main parachute, a vent is activated to equalize the interior and exterior pressure on the command module to insure pressure relief at landing impact. For normal flight this occurs at 10,000 ft. at an atmospheric pressure of 10 psi. For escape from the launch pad this occurs at the lower altitude of deployment.

3.33 Maximum Booster Shear and Bending Moment

The Saturn Booster is designed for the following maximum shears and bending moments at the spacecraft separation plane.

Saturn C-1 Vehicle

Maximum Transverse Shear = 17,700# (limit)

Maximum Bending Moment = 2.8×10^6 in# (limit)

(Condition: t = 55 sec., q = 5.34 #/sq. ft.,

= 5.6°, = 14°, 3 engine control)

Saturn C-2 Vehicle

Data not available at present.

3.4 Gust and Wind Shear Criteria

Since AMR is the only launching site considered for the spacecraft, then gust wind speed and wind shears utilized in design shall be for the immediate vicinity of Cape Canaveral, Florida. However, since aborted flight can cause a landing over a wide area of the earth, global gust and wind criteria shall be utilized in design for approach and landing conditions. Other limitations are applied to wind velocity and gusts while the spacecraft is on the launch pad.

3.41 Cape Canaveral Data, Ground and Boost Phase Gusts and Wind Shears

For launch and boost phase of flight, 2 probability data for the worst wind month at Cape Canaveral (Same as Saturn Criteria) shall be utilized for design. (Ref: George C. Marshall Space Flight Center Memorandum, E.D. Geiseler, Consolidated Wind Magnitude, Wind Shear, and Wind Speed Change Criteria for Saturn Vehicle Design, October 4, 1960). See attached copy.

3.411 Wind and Gust Velocities - On Launch Stand

The spacecraft shall be designed for a 60 ft/sec steady wind in combination with a ± 30 ft/sec horizontal step gust. For higher velocities, auxillary supports are assumed.

During final countdown, the maximum steady wind shall be 40 ft/sec in any direction in combination with a step gust of ± 20 ft/sec or as limited by Saturn booster design conditions.

3.5 Trajectory Data

(a) Ground launched earth orbit and lunar trajectories are planned on the basis of avoidance, where possible, of the Van Allen Radiation belts around the earth. Launches will be from AMR and take maximum advantage of the polar radiation windows upon exit and re-entry.

(b) Earth orbit trajectories for training flights are limited to orbit altitudes below the inner Van Allen belt.

(c) Earth orbit redezvous trajectories are planned as an item of future growth.

The following rajectory data is used for current spacecraft design.

3.81 Radiation Considerations TM 23-26

3.811 Radiation Environment in Space

The latest information available on the flux and energy spectrums of the various components of ionizing radiation shall be collected and charted to provide data on Earth-Lunar Space. Design criteria thus developed shall be submitted to NASA for approval for use in design of the spacecraft.

3.812 The following radiation components shall be considered:

Galactic Cosmic
- Protons
- Alphas
- Heavy Nuclei
- Earth albedo & secondaries
- Lunar albedo & secondaries

Solar Particles
- Protons
- Electrons
- Alphas
- Neutrons (If detected in the future, undetected as yet)
- Earth albedo and secondaries
- Lunar albedo and secondaries

Electromagnetic Radiation
- Solar primary spectrum
- Earth secondary spectrum
- Lunar secondary spectrum

Particles Trapped in Magnetic Field
- Van Allen belt protons
- Van Allen belt electrons
- Possible Lunar Belt particles (undetected as yet)

3.813 Solar flare activity and their effects on radiation levels

The possibility of short range prediction of solar flares shall be evaluated from available literature. (For the present prediction capability is assumed and mission planning is also assumed to coincide with periods of low probability of occurrence of major flares. Prediction of major flare activity would be grounds for scrubbing a flight.)

The change in flux and energy levels from ambient radiation levels will be evaluated from the latest available data on solar flare effects. The radiation level effects will be established versus flare magnitudes. The probability of encountering a given flare magnitude will be developed, consistent with prediction techniques.

3.814 Radiation Dose

A method of determination of radiation dose is outlined as follows:

1. Definition of flux and energy spectrum of components of radiation in space, including particulate and E/M.
2. Determination of mission exposure for each component of radiation including particulate and E/M.
3. Development of vehicle equivalent wall criteria for radiation effects and distributions in various vehicle zones.
4. Definition of vehicle wall modification and conversion of flux and energy spectrum, for each radiation component & secondaries.
5. Definition of biological effectiveness of flux and energy spectrums of each component of radiation.
6. Determination of dose from flux and energy spectrum of each component of radiation per vehicle zone.
7. Total biological dose for each vehicle zone by summation of individual component dose.

Since a large part of the data necessary for such an analysis does not exist in the technology at present, this method is considered more of a definition of a goal about which development can be concentrated. As better information on space environment, wall effects, and biological effects are developed, the analysis will become more realistic.

In the mean time, the limited information that is currently available from today's technology will be applied in the manner outlined to determine total biological dose. The total allowable radiation dose for the crew is currently limited to the following maximums.

1. Normal mission 5 REMS
2. Emergency condition 100 REMS

3.815 Radiation Protection:

The basic design philosophy to be followed to determine radiation protection requirements will be as follows:

1. Evolve the spacecraft design neglecting the effects of ionizing radiation.
2. Determine the radiation shielding effectiveness of the evolved design.
3. Determine methods and techniques that can be employed to minimize radiation inputs.
4. Determine 2σ probability biological dose expected for given space missions based on the best environmental data currently available.
5. Determine the requirements for radiation shielding and the resulting spacecraft modifications to stay within the specified dosage limits.

3.82 Meteoroid Criteria

3.821 Meteoroid Environment in Earth-Lunar Space.

The latest available data shall be utilized to chart the meteoroid particle size, flux, and velocity spectrum encountered in Earth-Lunar Space. This environmental criteria shall be submitted to NASA for approval prior to spacecraft design.

(Currently Whipple's 1957 tables are being utilized for preliminary design).

3.822 Mission Exposure to Meteoroids

In defining the mission exposure consideration will be given to the following:

Flight path in Earth-Lunar Space

Equivalent exposed surface area of critical systems.

The effect of Earth and Lunar shading when vehicle is in close proximity.

Accumulated exposure based on total time in space flight.

3.823 Meteoroid Penetration

A penetration model based on the latest available hypersonic impact data will be established and submitted to NASA for approval prior to spacecraft design.

(Currently Summers equation (NASA D-94) for determining penetration depth is being utilized for preliminary design)

Consideration will be made of the use of meteor bumper shields in determination of required material thickness to prevent penetration.

The overall design shall be based on a minimum probability of no complete penetration of 95%.

3.91 Metabolic Requirements

Basis for life support systems.

<u>Metabolic (75th percentile man)</u>

Metabolic heat rate	600 BTU/hr/man
O_2 consumption	2.367 #/man day
CO_2 production	2.8 #/man day
H_2O production	8.0 #/man day
respired/prespired	3.45 #/man day
urine	4.2 #/man day
feces	.35 #/man day
Food (assumed dry food ration with 7.5% water (95 Percentile man)	1.4 #/man day
Water (Drinking & food mixing & sanitation)	10.00 #/man day

Food for 3 men for 14 days is stored in the spacecraft

Water, 1 days supply for 3 men, is carried aborad. Continuous recycling of urine and absorbed moisture from the cabin air is performed to provide water of drinking purity.

3.92 Cabin Conditions

Cabin total pressure	(632 mm Hg, 12.2 psia) (equiv. to 5,000 ft. alt.)
Partial pressure	
O_2	159 mm Hg
Inert gas (N_2)	459 mm Hg
Water Vapor	11 mm Hg
	Equivalent to 50% rel. humidity at 68°F at sea level
CO_2 initial	.3 mm Hg
CO_2 max.	3.8 mm Hg
Trace gases	---
Air Temperature	70°F ± 10°F
Re-entry wall temp.	200°F max.
Air velocity	40 ft/min (max.)
Leakage	.05 #/hr. at 12.2 psia diff. pressure
Free ion limits in cabin air	Ion count ?

Air supply

2 complete charges from 5.5 psi up to 12.2 psi plus normal leakage and metabolic requirements for 14 days.

CO_2 Absorption reserve:

Recycle and ejection capability in mission module. 200% reserve of chemical absorber in re-entry vehicle. Cabin conditioning for 12 hr. post landing survival using ventilating fan to pull in outside air. Residual O_2 also available.

[redacted]

3.95 Cabin Air Contamination Limits

[illegible]

[illegible]	[illegible]	TLV in ppm
[illegible]	[illegible] Respiration	[illegible]
[illegible]	[illegible]	
[illegible]	Cooking	[illegible]
[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]
[illegible]	[illegible] propellant	25
[illegible]	[illegible]	[illegible]
[illegible]	Synthetic rubber	[illegible]
[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible] million particles/[illegible]
[illegible]	[illegible]	[illegible] million particles/[illegible]
	[illegible]	
	[illegible]	

[redacted]

3.93 (Continued)

Substance	Source	TLV in ppm
	Plastics	
	Metals	
Free ions	Irradiation of cabin air	Ion count?

[redacted]

4.2 [illegible]

4.11	[illegible]	ULT. FACTORS
4.111	[illegible] (Same as [illegible])	1.35
4.112	[illegible] ([illegible]) ([illegible])	1.25
4.113	[illegible] ([illegible])	1.5
4.12	[illegible]	
4.121	[illegible] ([illegible])	1.5
4.122	[illegible] ([illegible])	1.35
4.123	[illegible] ([illegible])	1.00
4.124	High Pressure [illegible] ([illegible] proof by test)	1.38
4.125	High Pressure [illegible] ([illegible] (4) on [illegible] life of system)	3.00
4.13	[illegible] ([illegible])	1.5

[redacted]

4.2 [illegible] Heatings

4.21 Reentry Heat Shield Factors:

Assuming use of both ablation and radiation heat shield types, the following heat pulse criteria are established as shown in Fig.

q_{Ult} = heat shield design flux at any point = q_H (Limit) x 1.25
= BTU/ft.^{2}/sec.

where: q_H = nominal limit input heat flux at any point on the reentry vehicle from reentry trajectory design conditions.

Distribution of q_{Ult} at any point varies with time, angle of attack and position on the reentry body such that the following heat pulse values are not exceeded.

4.211 Radiation shields, unaided by use of ablation coatings, shall be designed for a heat pulse in which the maximum flux does not exceed the following:

$$q_R = \sigma \epsilon (T_{Max})^4$$

Where: σ = Stefan-Boltzmann Constant
ϵ = Surface emissivity
T_{Max} = Allowable operating surface temperature of radiant shield

4.212 The heat pulse for design of ablation material extends to a minimum flux value with time, such that after depletion of ablation material, the heat flux to the radiation shield under the ablation material will be no greater than:

$$q_{R_1} = \sigma \epsilon (T_{Max} - 200°F)^4$$

4.22 Temperature resulting from q_{Ult} shall be combined with limit loads or ultimate loads shall be combined with temperatures resulting from nominal heat flux, q_H, whichever is the most critical.

4.23 Understructure such as the cabin pressure wall shall be limited to a design maximum temperature of 200°F.

4.24 Miscellaneous Heat Fluxes

$\left.\begin{array}{l}\text{Ascent heating flux} \\ \text{Solar heat flux} \\ \text{Earth shine \& moon shine flux}\end{array}\right\}$ Ultimate factor = 1.00

4.3 Design Allowable Criteria - Structural Materials

4.31 Metallic Materials Applications:

Three distinct thermal usage ranges shall be considered in determining metallic material allowables for structural design. The following sketch shows a typical short time tensile strength-temperature diagram in which these ranges have been superimposed.

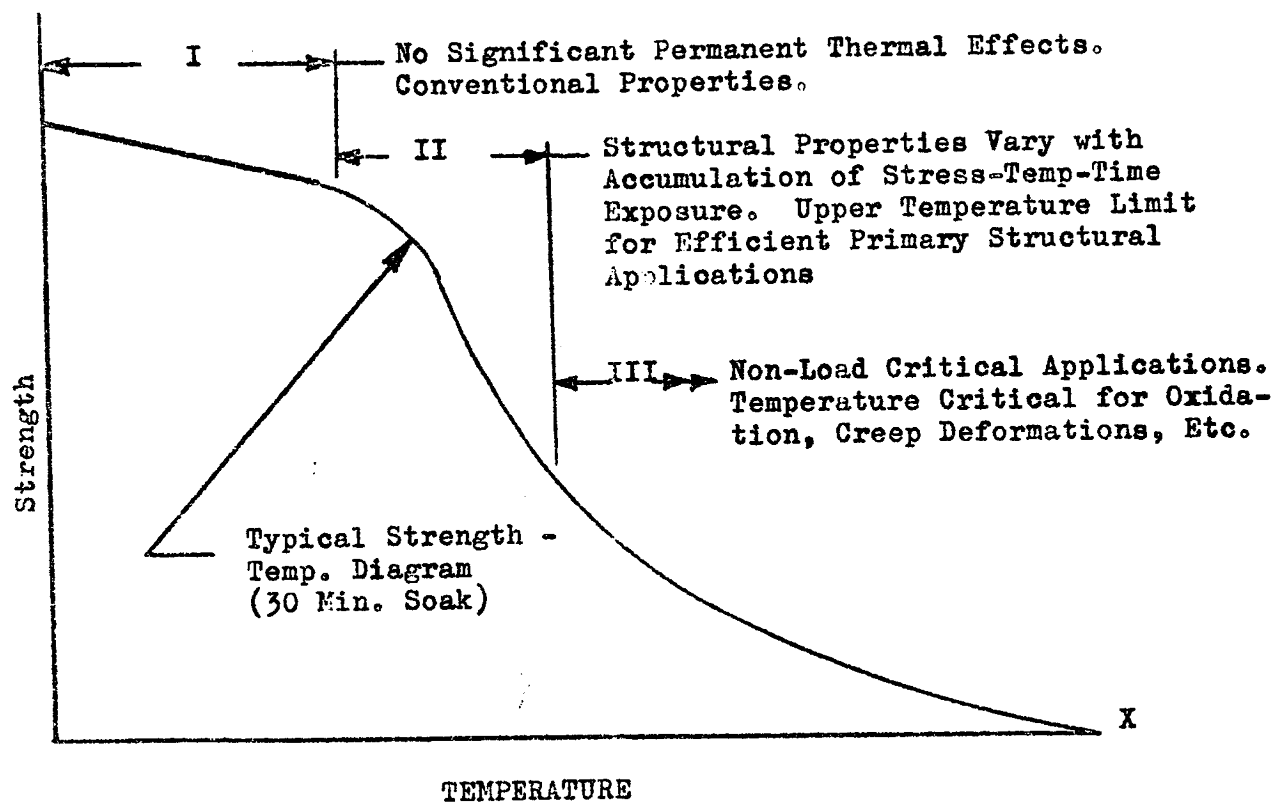

FIG. 4.311 TEMPERATURE RANGES FOR METALLIC MATERIAL DESIGN APPLICATIONS

The determination of design allowables for metallic materials applications in each range are outlined in the following paragraphs.

4.32 Temperature Range I:

4.321 Within this temperature range the basic properties of metallic materials are essentially stable, and can be considered to vary slightly with level of temperature. However, for design applications, no permanent thermal effects are considered. Standard material properties data may be utilized for static design.

4.322 The upper limit for this range is the temperature where thermally induced changes in material properties become significant for design. An approximation of this upper limit may be made by taking the temperature where the conventional 30 minute soak strength-temperature curve initially begins to change shape rapidly as indicated in Fig. 4.311.

4.333 Sources of material properties data are manufacturers minimum guaranteed properties, Mil HDBK-5, the Martin Structural Design Manual, or minimum properties derived from sufficient specific tests to establish 3σ probability values.

4.33 Temperature Range II

4.331 Within this temperature range, metallic materials are still considered for primary structural applications. Properties vary with temperature level and with the degree of prior exposure to a stress-temperature-time environment. The definition of the upper limit of temperature for this range is rather arbitrary. The main point to consider is structural efficiency. This can be influenced by either the creep rate, the strenght/density or modulus/density ratios for a given material. If these exceed or drop below required values as established for a given application, then the material would no longer be considered for the application.

The temperature at which this transition ocurrs would be considered the upper limit for the temperature range.

4.332 The difficulty in pinning down a finite value for this temperature limit is characterized by the varied requirements and limitations for different design applications. By inspecting available test data on any given material, a sort of rationalized temperature limit can be arrived at, that falls somewhere on the steep slope of the strength-temperature curve, Fig. 4.311.

4.333 The problem of predicting material properties in this temperature regime, with suitable accounting for the effects of prior history, has been approached in many ways. To date, no reliable methods have been evolved that enable prediction of material properties exposed to complex environments.

For a given design problem, it is necessary to establish allowable properties for materials. The variable properties in this temperature regime have led to gross confusion or unenlightened approximation which is not compatable with requirements for minimum weight and highly reliable structure for the spacecraft. More definitive methods of determining material allowables for complex stress-temperature-time interactions are needed without the complication of duplication of the environment in material tests.

If certain design assumptions are acceptable for the detail design of specific spacecraft structure, then the method outlined on the following pages can be utilized to yield reliable design allowables.

These design assumptions are as follows:

1. Minimum material properties ocurr at the maximum structural temperature encountered with suitable prior history accounted for.

2. Creep in the composite spacecraft structure will not be significant due to use of design factors of safety, choice of materials with high structural efficiency, and due to the low period of time under the environmental extremes of stress and temperature normally experienced by the spacecraft.

3. The structure can be considered critical for peak environmental conditions encountered.

4. Compression properties are assumed the same as tensile properties for all structural metals.

4.334 The effect of prior history of exposure on the properties of materials can be determined in a reliable manner by an adaption of a method of random statistical sampling as reported in WADD-TR 60-777. The total population of variables and variable interactions (stress-temperature time) can be sampled with comparitively few tests. The scatter band of properties can be determined, as influenced by prior history of exposure over the whole black of environmental variables. Minimum properties derived from these scatter bands can then be utilized as design allowables, with the assurance that the environmental degradation will not exceed these minimums within the temperature regime.

The key to the random method is the technique for evaluating representative environmental change in properties. This is done by first separating the parameters into increments of stress, temperature, and time. Then applying these in random combination on specimens selected at random, such that all tests are run for identical times. The sequence of application of basic combination step functions, and the final static test temperatures are also chosen at random. By means of this randomization, the material property data thus derived should be a representative random sample spanning the entire range of environmental effects within the regime investigated.

4.335 Table 4.3351 shows a complete random sampling schedule for any material within a critical time dependent environmental range. 32 separate specimens are utilized to sample a factorial of 64 variable combinations. See Fig. 4.3352. By applying these in four separate sequences, a total possible population of 64^4 (16,777,216) variations are sampled.

The basic problem of validity of the data lies in complete randomization without introduction of bias. The tests must be conducted as accurately as possible exactly as the schedule indicates. Each number test must be treated individually in sequence. Attempts at grouping for the same temperature or other methods to shortcut time should be avoided as they would tend to bias the results.

The total time for each test is the same. A value must be chosen that is compatible with inducing significant degradation in the material from the standpoint of overaging and straining within the temperature regime. This time increment and the corresponding range of stress should be selected such that the worst combination of variables would be sufficiently severe to

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R.R. DRUMMOND
3-7-61

FIGURE 4.3361

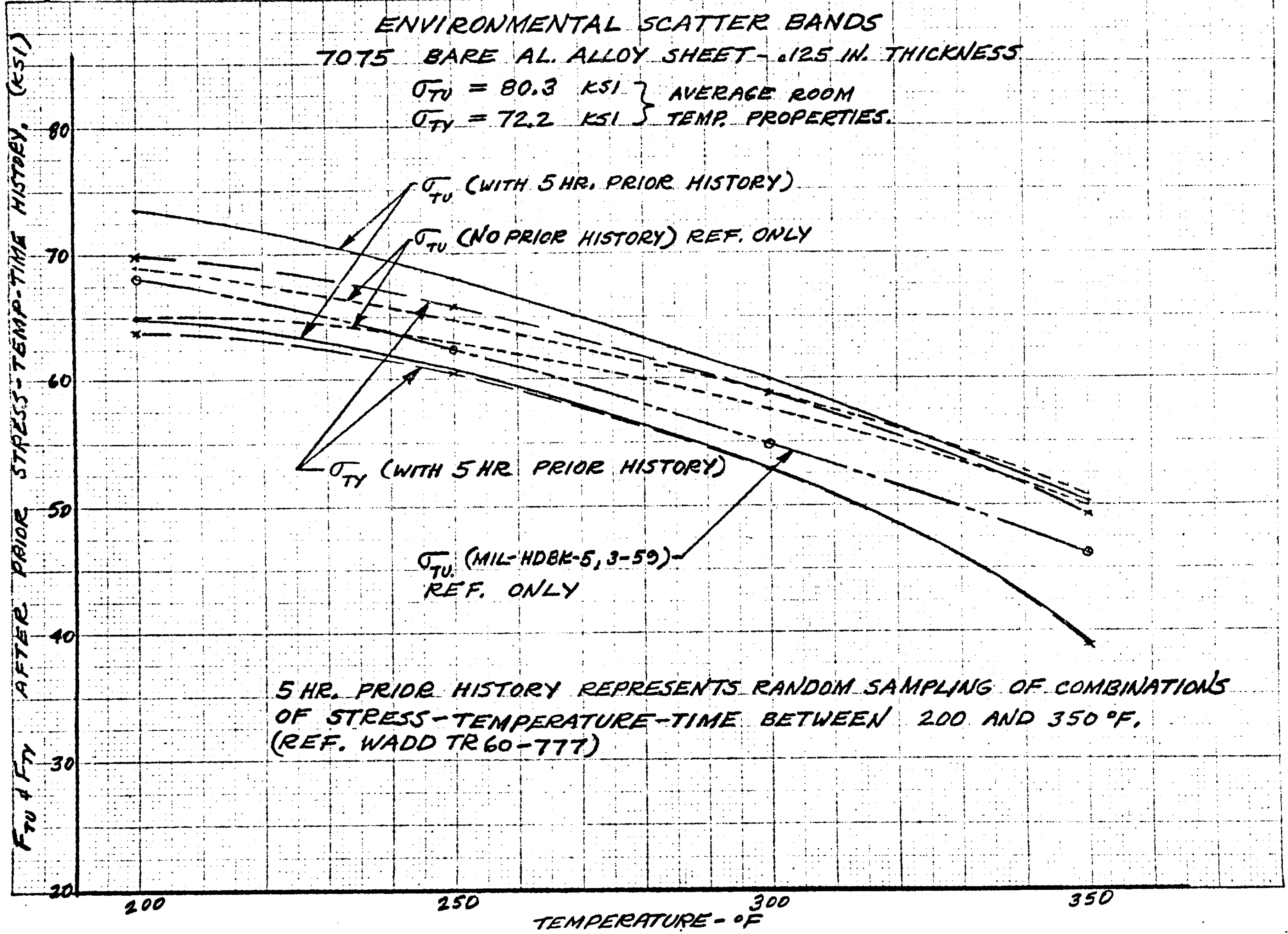

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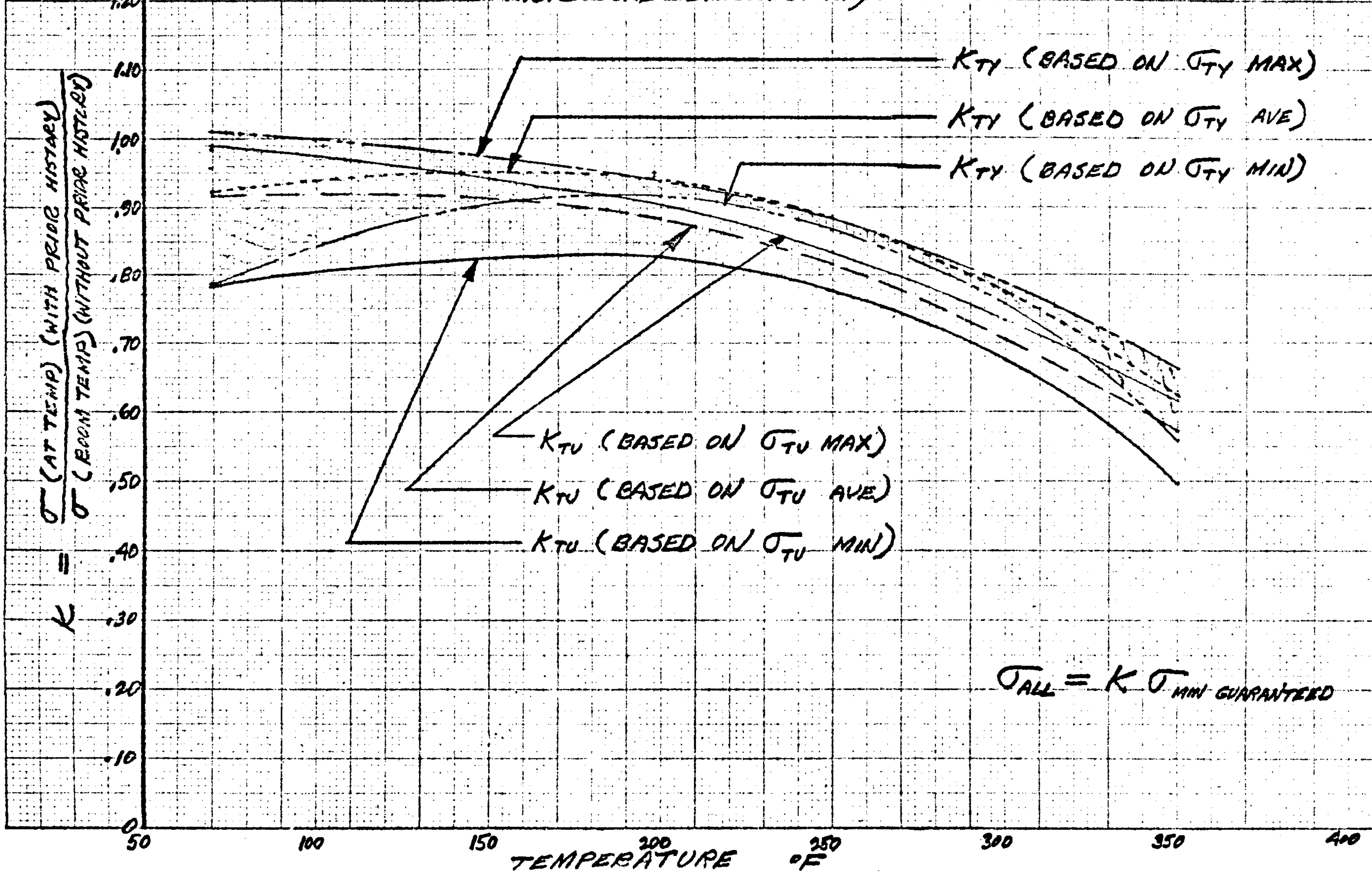
FIGURE 4.3362

ENVIRONMENTAL K FACTORS

7075-T6 AL. ALLOY BARE SHEET-.125 IN. THICKNESS
(INCLUDING EFFECTS OF
PRIOR LOAD-TEMP HISTORY)



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[redacted]

TABLE 4.3551

STRESS - TEMPERATURE - TIME EFFECTS

GENERALIZED RANDOM SAMPLING SCHEDULE FOR MATERIALS

	Static Test Temperature (After Exposure)	Test Number	Specimen Number	Random Step Function Sequence (Total Test Time Constant)
1	T_1	1	S-20	$\sigma_1 T_2 t_1 + \sigma_3 T_4 t_4 + \sigma_1 T_3 t_3 + \sigma_4 T_3 t_2$
2		5	S-11	$\sigma_3 T_1 t_1 + \sigma_3 T_3 t_3 + \sigma_2 T_2 t_4 + \sigma_2 T_2 t_2$
3		8	S-3	$\sigma_4 T_4 t_1 + \sigma_1 T_1 t_2 + \sigma_4 T_2 t_3 + \sigma_1 T_3 t_4$
4		13	S-4	$\sigma_3 T_2 t_2 + \sigma_1 T_1 t_4 + \sigma_4 T_4 t_1 + \sigma_1 T_2 t_3$
5		22	S-30	$\sigma_3 T_3 t_3 + \sigma_3 T_3 t_1 + \sigma_2 T_4 t_2 + \sigma_1 T_1 t_4$
6		25	S-21	$\sigma_1 T_1 t_4 + \sigma_2 T_3 t_3 + \sigma_3 T_1 t_1 + \sigma_1 T_3 t_2$
7		30	S-19	$\sigma_3 T_4 t_4 + \sigma_2 T_2 t_2 + \sigma_4 T_4 t_3 + \sigma_1 T_2 t_1$
8		31	S-13	$\sigma_4 T_1 t_4 + \sigma_3 T_2 t_2 + \sigma_3 T_3 t_1 + \sigma_3 T_1 t_3$
9	T_2	3	S-16	$\sigma_2 T_1 t_1 + \sigma_4 T_2 t_3 + \sigma_1 T_3 t_2 + \sigma_4 T_3 t_4$
10		7	S-28	$\sigma_4 T_2 t_1 + \sigma_1 T_3 t_4 + \sigma_1 T_1 t_2 + \sigma_4 T_2 t_3$
11		10	S-24	$\sigma_1 T_3 t_2 + \sigma_1 T_2 t_3 + \sigma_4 T_3 t_4 + \sigma_3 T_1 t_1$
12		21	S-23	$\sigma_3 T_1 t_3 + \sigma_4 T_3 t_4 + \sigma_3 T_4 t_2 + \sigma_3 T_3 t_1$
13		23	S-31	$\sigma_4 T_2 t_3 + \sigma_2 T_1 t_1 + \sigma_2 T_4 t_4 + \sigma_2 T_4 t_2$
14		26	S-12	$\sigma_1 T_3 t_4 + \sigma_1 T_3 t_2 + \sigma_3 T_1 t_3 + \sigma_4 T_4 t_1$
15		27	S-18	$\sigma_2 T_2 t_4 + \sigma_4 T_4 t_1 + \sigma_2 T_1 t_3 + \sigma_4 T_1 t_2$
16		28	S-6	$\sigma_2 T_4 t_4 + \sigma_1 T_2 t_1 + \sigma_2 T_2 t_2 + \sigma_2 T_3 t_3$
17	T_3	6	S-14	$\sigma_3 T_3 t_1 + \sigma_4 T_1 t_2 + \sigma_1 T_4 t_3 + \sigma_3 T_4 t_4$
18		11	S-27	$\sigma_2 T_2 t_2 + \sigma_1 T_4 t_1 + \sigma_3 T_3 t_3 + \sigma_2 T_4 t_4$
19		15	S-8	$\sigma_4 T_1 t_2 + \sigma_4 T_1 t_4 + \sigma_1 T_4 t_1 + \sigma_1 T_4 t_3$
20		16	S-22	$\sigma_4 T_3 t_2 + \sigma_2 T_1 t_3 + \sigma_3 T_4 t_4 + \sigma_2 T_1 t_1$
21		17	S-7	$\sigma_1 T_2 t_3 + \sigma_2 T_4 t_2 + \sigma_1 T_2 t_1 + \sigma_4 T_1 t_4$
22		19	S-17	$\sigma_2 T_1 t_3 + \sigma_2 T_2 t_4 + \sigma_2 T_1 t_1 + \sigma_3 T_4 t_2$
23		24	S-5	$\sigma_4 T_4 t_3 + \sigma_3 T_2 t_4 + \sigma_4 T_2 t_1 + \sigma_1 T_1 t_2$
24		32	S-29	$\sigma_4 T_3 t_4 + \sigma_3 T_1 t_3 + \sigma_2 T_3 t_2 + \sigma_4 T_2 t_1$
25	T_4	2	S-9	$\sigma_1 T_4 t_1 + \sigma_2 T_4 t_4 + \sigma_4 T_1 t_2 + \sigma_3 T_3 t_3$
26		4	S-15	$\sigma_2 T_3 t_1 + \sigma_4 T_3 t_2 + \sigma_3 T_2 t_4 + \sigma_2 T_1 t_3$
27		9	S-32	$\sigma_1 T_1 t_2 + \sigma_1 T_4 t_3 + \sigma_1 T_3 t_4 + \sigma_1 T_4 t_1$
28		12	S-25	$\sigma_2 T_4 t_2 + \sigma_4 T_4 t_3 + \sigma_4 T_1 t_4 + \sigma_2 T_3 t_1$
29		14	S-10	$\sigma_3 T_4 t_2 + \sigma_2 T_3 t_1 + \sigma_2 T_3 t_3 + \sigma_3 T_2 t_4$
30		18	S-1	$\sigma_1 T_4 t_3 + \sigma_4 T_2 t_1 + \sigma_1 T_1 t_4 + \sigma_3 T_2 t_2$
31		20	S-2	$\sigma_2 T_3 t_3 + \sigma_3 T_4 t_2 + \sigma_2 T_3 t_1 + \sigma_2 T_2 t_4$
32		29	S-26	$\sigma_3 T_2 t_4 + \sigma_3 T_1 t_1 + \sigma_3 T_2 t_2 + \sigma_4 T_4 t_3$

$\left.\begin{array}{l} \sigma = \text{Stress Level} \\ T = \text{Temperature Level} \\ t = \text{Time Increment} \end{array}\right\}$ 4 Different Values for each Input Variable.

Half factorial sampling of 64 possible combinations of input variables.

Total Number of Tests = 32, Total Number of Specimens = 32

Total Environmental Population Sampled = 64^4 = 16,777,216.

[redacted]

FIGURE 4.3352

SELECTED TEST COMBINATIONS

(HALF REPLICATE FACTORIAL)

TIME HRS.	TEMPERATURE °F				STRESS PSI
	T_1	T_2	T_3	T_4	
t_1		X		X	σ_1
	X		X		σ_2
	X		X		σ_3
		X		X	σ_4
	X		X		σ_1
t_2		X		X	σ_2
		X		X	σ_3
	X		X		σ_4
t_3		X		X	σ_1
	X		X		σ_2
	X		X		σ_3
		X		X	σ_4
t_4	X		X		σ_1
		X		X	σ_2
		X		X	σ_3
	X		X		σ_4

X denotes selected combination of $\sigma T t$ in order to sample the total factorial of 64.

 [redacted]

induce a few premature failures. This insures that degradation of the material extends up to a maximum value. The economic factors for the tests dictate that this be accomplished in the shortest practical test time.

An estimate of total test time and maximum stress level can be made from constant load-constant temperature creep data. A given deformation limit can be set up as the criteria of failure and stress-time maximum relationships determined. The maximum stress level should be no higher than that expected in realistic design applications. If this criteria, is followed, the environmental step function tests would be expected to be slightly more critical in their extreme ranges.

Data to be recorded during the step function tests are total deformation and creep. Accurate stress strain diagrams and ultimate strengths are to be recorded for the tests after prior history. Other basic parameters such as elongation, thermal expansion poissons ratio and reduction of area should also be recorded.

4.336 Figure 4.3361 shows actual test data generated from 5 hour tests of 7075-T6 bare sheet where the stress level varied between 70 to 85% of yield (at temperature) (Ref. WADD TR60-777). Inspection of the curves show that the environmental step function test data forms a scatterband that encompasses data from specimens having no prior history as well as elevated temperature strengths presented in MilHDBK-5. This environmental data represents a purely random sample across the environmental range spanned by the tests. It further represents the theoretical maximum range of environmental degradation for the particular material,

[redacted]

since a few premature failures of specimens occurred during the test program due to extreme combinations of parameters. The values of the data generated pertain to the test material only.

If this data is further generalized into non-dimensional form, it can be assumed to apply more generally to the given class of material. Divide the environmental property data scatter band extremes by the property data at room temperature to develop a non-dimensional factor K_σ which represents the percent degradation due to environmental effects. A plot of K_σ vs temperature, Fig. 4.3362 shows a scatter band that can be assumed to apply generally to the form of material tested. (More substantiation by test should be made of the validity of this assumption for the 7075-T6 material). This is the type of information that can be catalogued for each material to determine basic allowable properties for design. It is also noted that these data are independent of the design application.

4.337 By utilizing the following relationship,

$$\sigma_{ALLOWABLE} = K_\sigma \, \sigma_{\substack{MIN. GUARANTEED \\ AT ROOM TEMP.}}$$

allowable strength can be determined for a material exposed to a design environment within the given temperature regime.

All that is necessary is to determine the peak temperature to be reached for the particular design application, determine K_σ from the curve for the material and multiply this times the standard minimum guaranteed properties at room temperature.

[redacted]

[illegible] values can be obtained for yield and modulus from the same test data. The stress strain curves recorded during the tests can be plotted and grouped according to test temperature and ultimate strength such, that interpolation can be made between curves for design.

4.338 Creep data generated during the environmental test cannot be as readily generalized since this measurement is very sensitive to material processing history. Duplicate tests will correlate only if the specimens are taken side by side in the same sheet material. However, the data can be studied from the standpoint of discrete combinations of variables that cause rapid creep under varying sequences of application. These ranges are usually avoided in design since they generally are composed of combinations in the upper extreme ranges of variables. The normal use of safety factors would depress the working stress in the structure to a point below the critical range.

Below this critical range of parameters a fairly orderly relationship exists that somewhat parallels the strain hardening rule for creep. In WADD TR60-177, a modification of this rule is presented in the form of changing to apparent temperatures for identical step functions appearing in later sequences in the tests. The recorded creep for each succeeding step is taken in an equivalent time increment, and compared, by Larson Miller parameters, to one another on the basis of equivalent changes in temperature necessary to produce the differences in the Larson Miller parameter. These apparent temperatures are then used in place of the actual steps for strain hardening rule calculations. This curve fitting technique can be utilized for design applications as long as the data are below the critical parameter range.

[redacted]

9) [illegible] temperature engines are [illegible] applications, where maintaining [illegible] is more important than the load carrying ability of the structure. However, deformation due to high creep rate [illegible] a problem in holding a desired shape.

10) [illegible] testing [illegible], [illegible], [illegible] design conditions. [illegible] the actual conditions expected of in usage.

[illegible]

[illegible] change in velocity [illegible] operation of the [illegible]. Except for the initial [illegible] velocity into lunar trajectory, the [illegible] energy shall be provided by onboard [illegible].

[illegible]

[illegible]

[illegible]

[illegible]	=	[illegible]
[illegible]	=	[illegible]
[illegible]	=	[illegible]

[illegible] corrections

[illegible]	=	[illegible]
[illegible]	=	[illegible]
[illegible]	=	[illegible]
[illegible]	=	[illegible]

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George C. Marshall Space Flight Center
Huntsville, Alabama

MEMORANDUM

To Distribution Date October 4, 1960

From Director, Aeroballistics Division

Subject Consolidated Wind Magnitude, Wind Shear, and Wind Speed Change Criteria for Saturn Vehicle Design

1. This memorandum describes the wind environment for which the Saturn vehicle shall be designed to function. A characteristic difference in the design viewpoint regarding wind criteria for which the Saturn vehicle shall be designed, as compared to a tactical military missile, is the relaxation of the wind magnitude which the system is expected to encounter. This philosophy assumes that the more extreme wind magnitudes can be avoided and wind levels up to the 95% probability-of-occurrence level (two-sigma) may be considered as extremes to be encountered by the vehicle. The wind criteria outlined in this memorandum provide for an operational capability 95% of the time for the worse wind month of the year at Cape Canaveral (Atlantic Missile Range), Florida. Therefore, this design condition provides for vehicle operations on an average for all but 1½ days for the worse wind month of each year. For other months of the year a higher probability for successful operations, relative to wind restrictions, will exist.

2. The stated wind environment conditions are presented in order to consolidate previously published information in one reference to insure coordination within all organizations involved in the Saturn Program. Although based on the best available data and design philosophy, the criteria presented are subject to revision based on the results of future investigations.

3. Wind Magnitude - Figure One provides data on the wind speed as a function of altitude. This profile is an envelope of the wind speeds at each altitude level associated with the 95% probability (two-sigma) level.

4. Wind Shear - Figure Two contains information on the wind shear spectrum associated with differentials in altitude from 100m to 4000m over the altitude range from 6 km to 18 km. These data are to be combined with the wind magnitudes in Figure One to establish wind build-up curves leading to the various wind magnitudes or fall-off rate from the given wind magnitudes.

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SUBJECT: Consolidated Wind Magnitude, Wind Shear, and Wind Speed Change Criteria for Saturn Vehicle Design

This procedure enables the association of small and large scale wind shears with the wind design magnitude. An example of this is presented in Figure Four. Also, see Reference 4 for further description of this procedure.

5. <u>Wind Speed Change</u> - Figure Three is simply a conversion of the data presented in Figure Two into another form. It should be noted that the wind change (shear) values represent the approximate 99.7 (three-sigma) level wind change (shear) to be expected over the indicated scale-of-distances (differentials in altitude) for the altitude range from 6 km to 18 km. They are to be associated with the wind magnitudes given in Figure One which for a given altitude level provides the wind magnitude. Figure Three will provide for any scale-of-distance between 100m and 4000m the approximate 99.7 (three-sigma) level expected wind speed change rate leading up to the wind magnitude or fall-off rate from the wind magnitude with increase in altitude.

6. <u>Example of Synthetic Wind Profile Construction</u> - Figure Four illustrates the employment of the data given in Figures One, Two and Three to combine the wind magnitude and wind shear. The profile is constructed on the assumption that the wind build-up conditions over the 100, 500, 1000 and 2000 meters scale-of-distances (differential altitude) below 12 km and the wind magnitude at 12 km are of interest to the problem under study. Reference is made to Figure One. The wind magnitude at 12 km is given as 75 m/sec. The wind build-up condition is obtained from Figures Two and Three as:

h(m)	w/ h (sec^{-1})	w (m/sec)
100	0.090	9
500	0.050	25
1000	0.035	35
2000	0.021	42

Figure Four illustrates the construction of a wind profile envelope building up to the 12 km wind magnitude as determined by the above shears. It should be noted that the wind speed does not build-up from zero in a step function at 10 km but builds up at a minimum slope from the surface.

7. The wind criteria presented herein should be applied to the vehicle without regard to direction, i.e., considered as wind from all compass directions relative to the vehicle. Furthermore, these wind criteria, based on our present knowledge of the wind structure over Cape Canaveral, provide for a 95% probability of operation during the worse wind month. The worse wind month is defined as the monthly period exhibiting the highest average wind speeds in the 10-14 km altitude region.

E. D. Geissler
Director, Aeroballistics Division

~~CONFIDENTIAL~~

REFERENCES

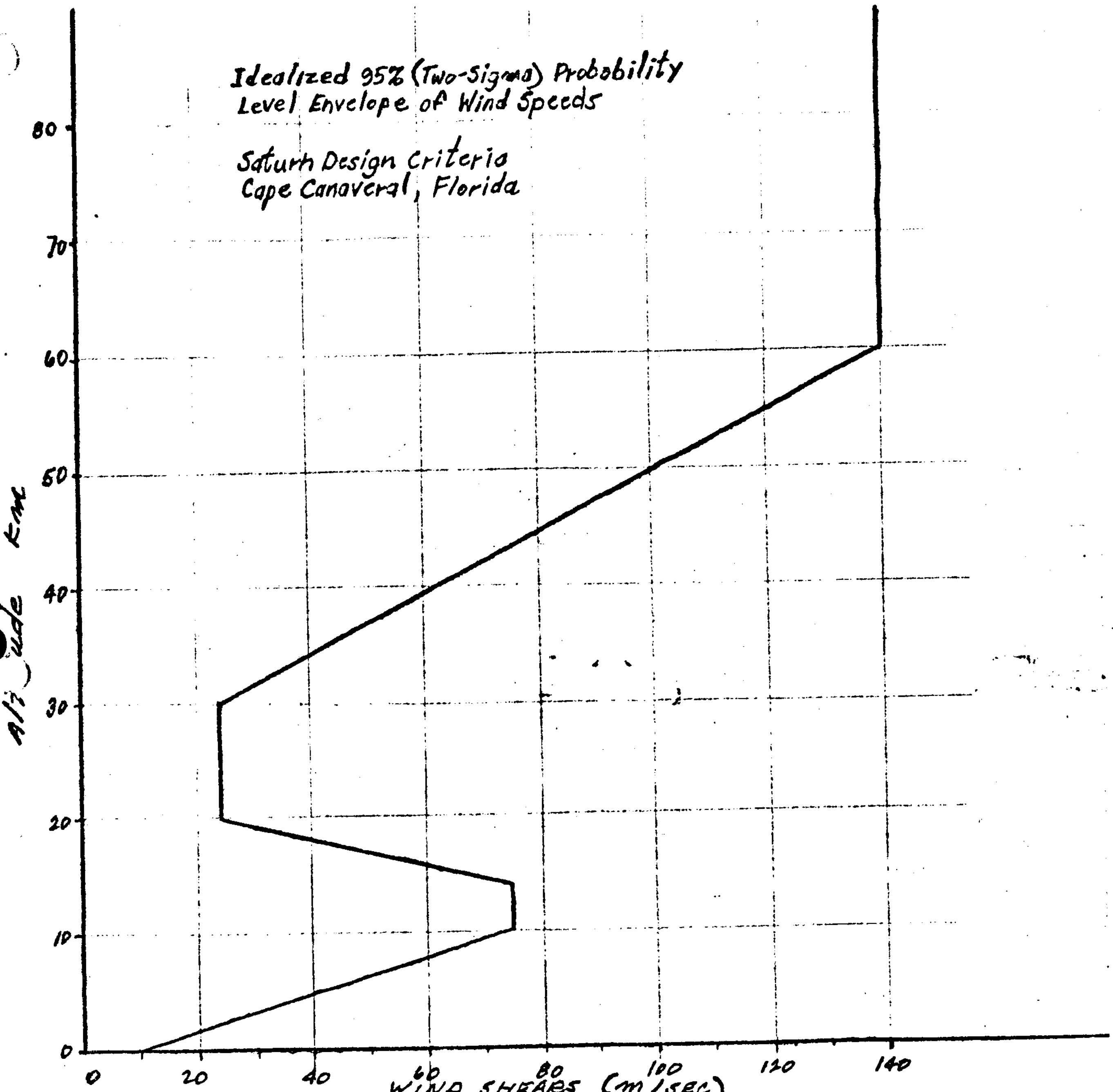
1. Vaughan, W. W.; Investigation of the Cape Canaveral, Florida, Wind Magnitude and Wind Shear Characteristics in the Ten to Fourteen-Kilometer Altitude Region, August 9, 1960, NASA-TN-D556.

2. Vaughan, W. W.; Idealized Wind Profiles Applicable for the Study of Vehicle Performance Between 50 km and 80 km Altitude, Cape Canaveral, (Atlantic Missile Range), Florida, August 1, 1960, NASA-TN-D560.

3. Vaughan, W. W.; Surface Wind Distribution for Cape Canaveral, Florida, (U) March 21, 1958, ABMA-DA-TN-103-58 (Confidential)

4. Sullivan, E. L., and Jean, O. C.; Saturn C-1 Control Study for Four Control Motors Operative (U), April 14, 1960, ABMA-DA-TM-30-60 (Secret)

5. Vaughan, W. W.; Analysis of Discrete Atmospheric Gust Velocity Data for Use in Missile Design and Performance Studies, November 20, 1959, ABMA DA-TR-68-59.



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Marshall Space Flight Center
Sept. 20, 1960

FIGURE ONE

[redacted]

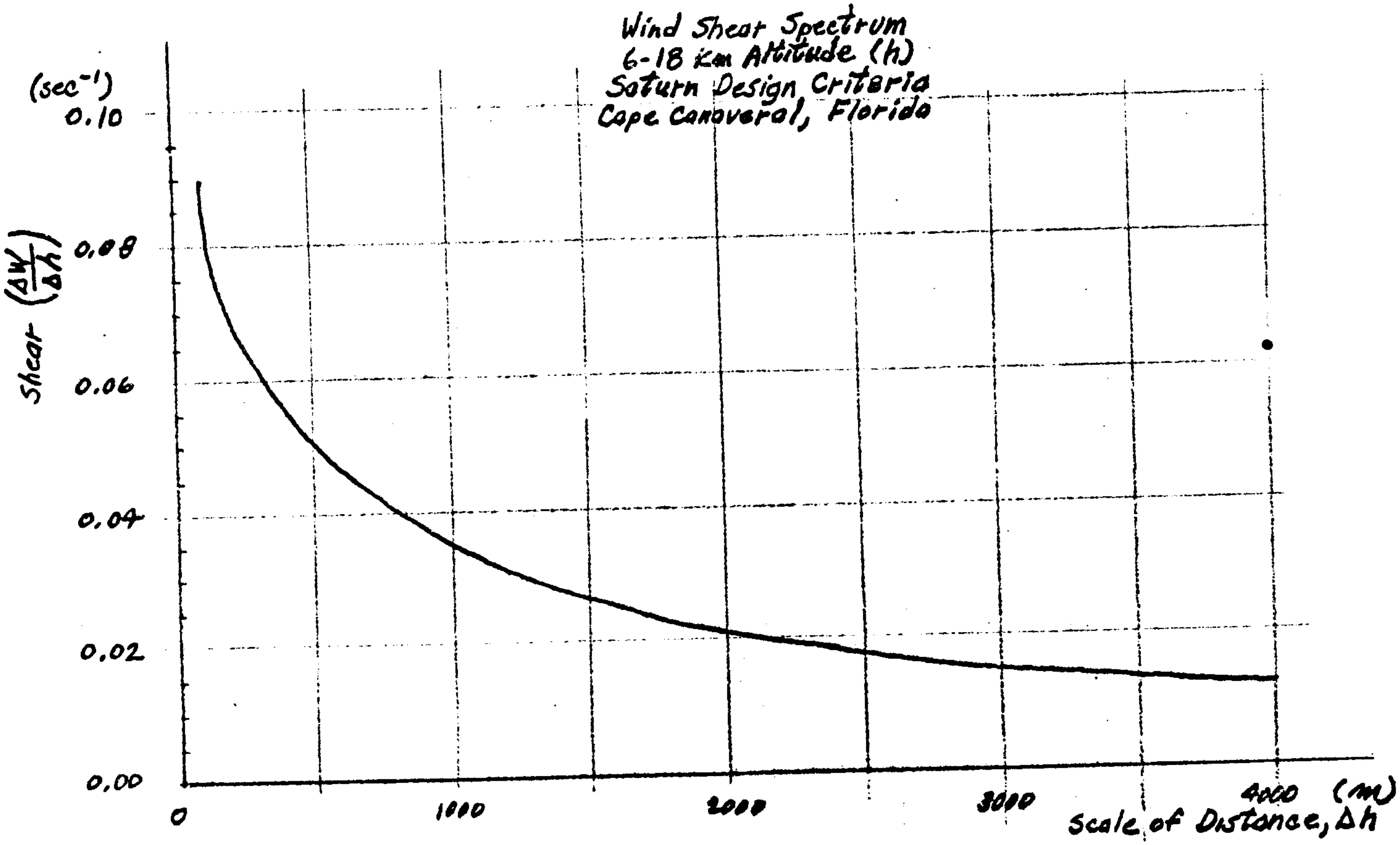

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FIGURE TWO.

[redacted]

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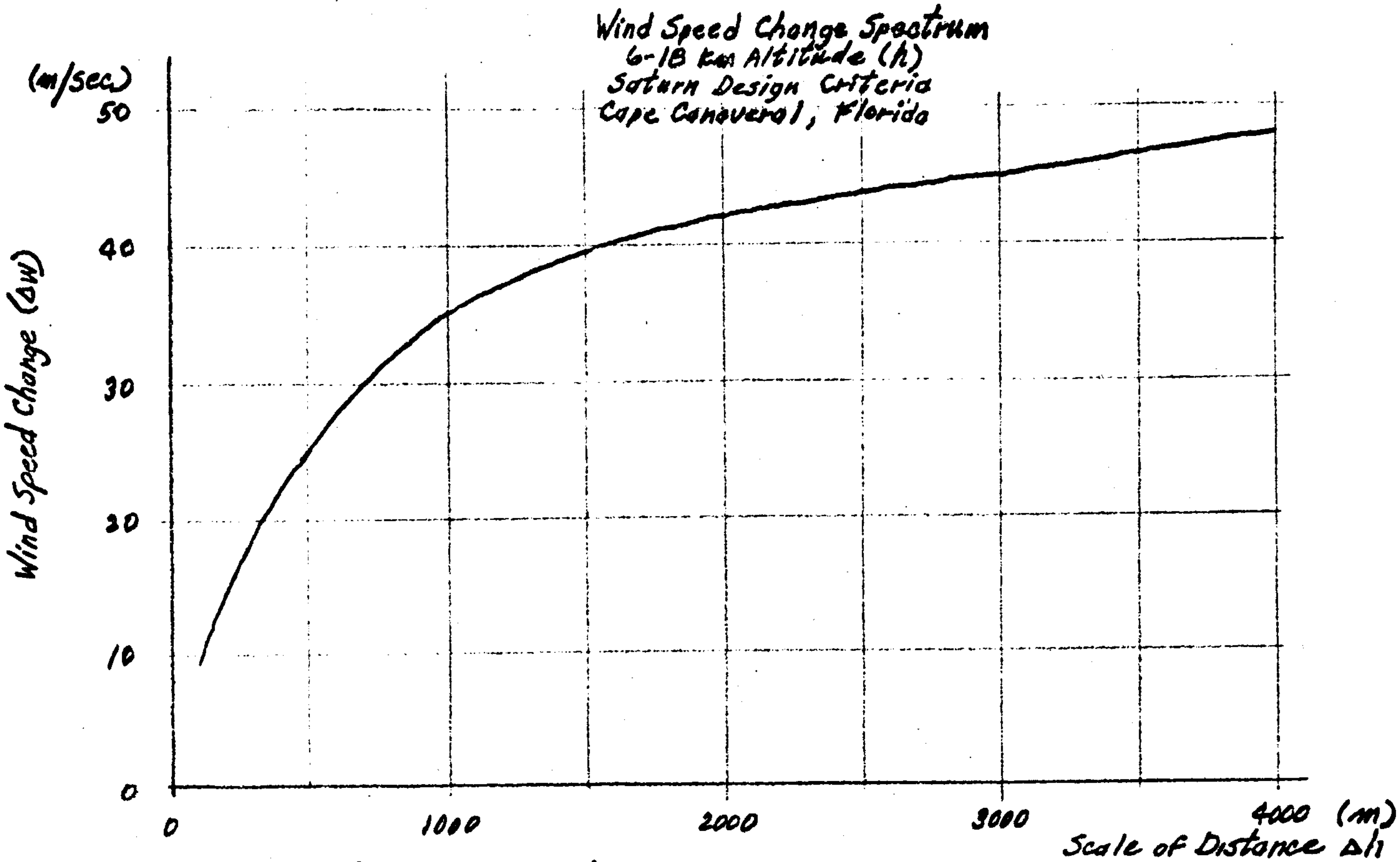

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Aeroballistics Division
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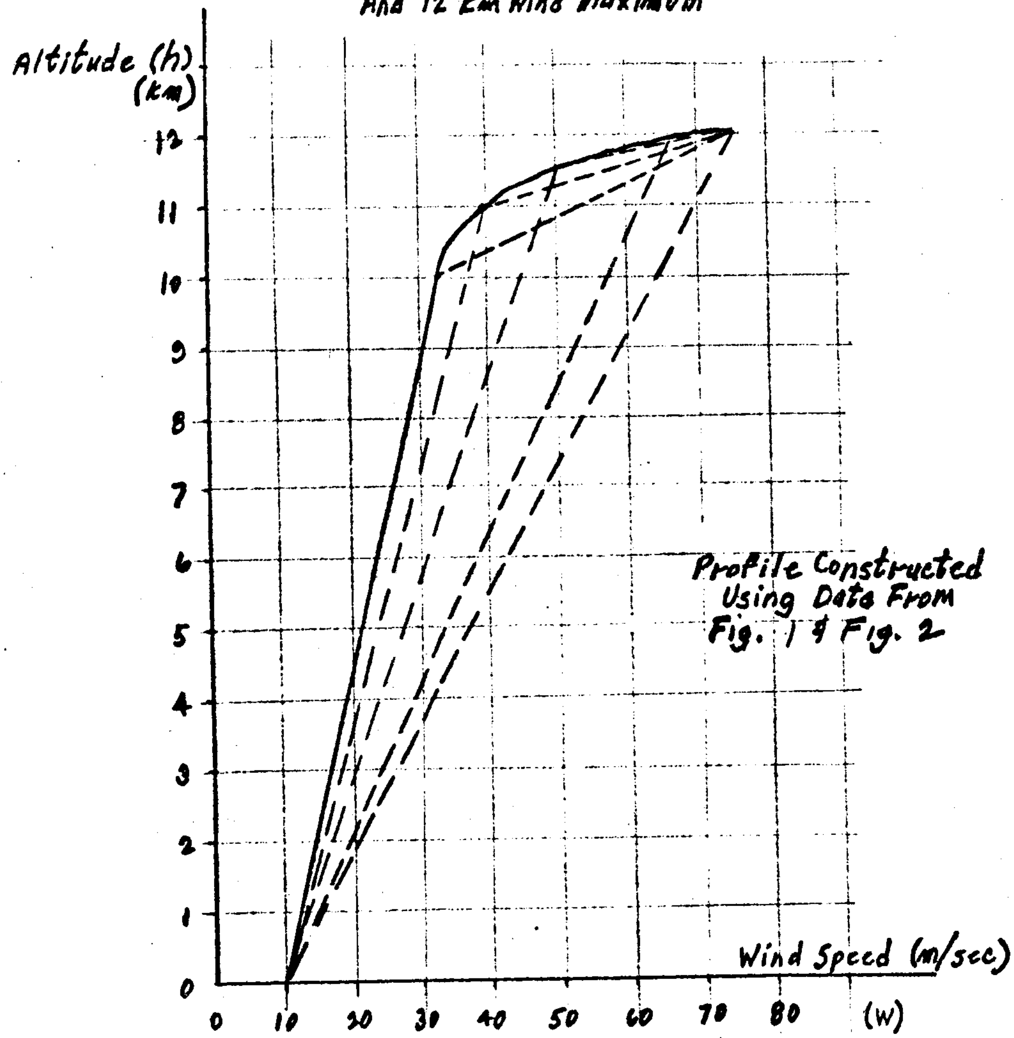
Sept. 20, 1960

FIGURE THREE

CONFIDENTIAL

Example of Synthetic Wind Profile Construction
Using 100m, 500m, 1000m & 2000m Wind Shear Values
And 12 km Wind Maximum



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Sept. 20, 1960.

[redacted] FIGURE FOUR

Design

TM-24

Technical Memorandum 24

Apollo Design

In the early phase of this program a wide variety of design concepts were investigated and evaluated. Both modular and integrated concepts were considered. Reentry vehicle types varied from ballistic, to lifting body, to flyable, designs. Various mission module types and many propellant tankage arrangements were studied. The attached drawings are typical of this work, and represent most of the concepts investigated.

The breadth of the study program was then narrowed to exclude all but the L-2C and W-1 reentry vehicle types. Each of these vehicles was then investigated in greater detail, in both modular and integrated form.

Integrated Configurations

The two integrated configurations are similar in many respects; the type of reentry vehicle employed being the most obvious difference. Differences in heat shielding, structure, and overall geometry also exist, however.

Each configuration includes a multiple nozzle, truss mounted, solid propellant escape rocket. Each reentry vehicle is sized to contain six persons and the equipments required to perform the mission. Propellant tankage consists of a conical oxidizer tank plus a toroidal fuel tank in each case.

An aft structural shell provides support for the tankage, and also serves as a meteorite bumper. External equipments, such as space radiators, solar cell array, and communication antennas are also supported by this structure.

[redacted]

For circum-lunar, and lunar orbit missions, three (3) personnel stations and the air lock would be replaced by the desired scientific gear and the additional life support equipment required.

Modular Configurations

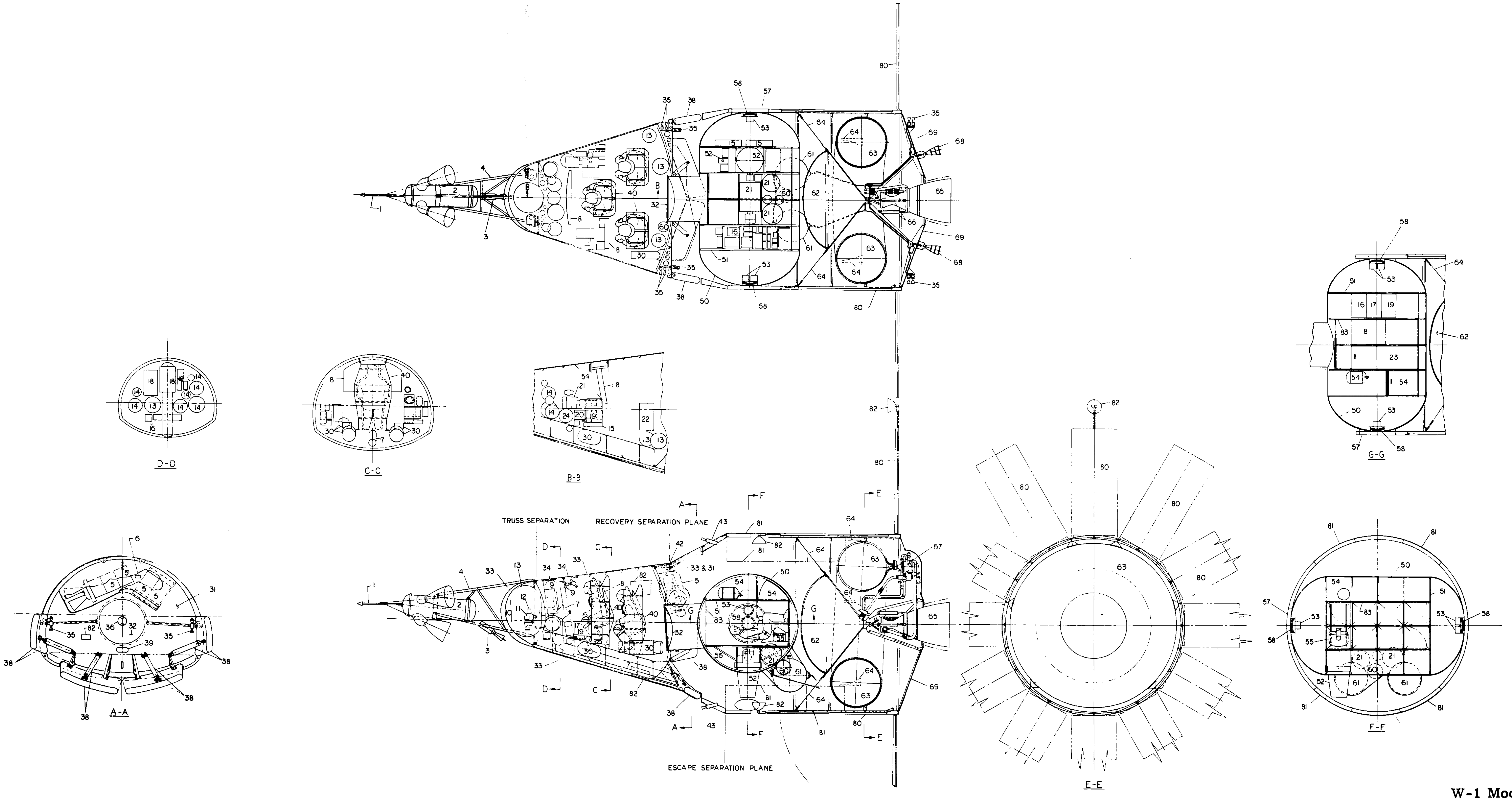
The two modular configurations are also very similar to each other, except for differences due to use of the basically different reentry bodies.

Escape provisions are the same in concept as described for the integrated configurations. Both reentry vehicles are sized for a crew of three (3) and appropriate equipments. The Mission and Propulsion modules, as well as the aft portion of the structural shell is identical for each configuration.

The Mission Module contains scientific equipment, living facilities (except sleeping quarters) and other gear not essential to reentry and post-reentry phases of the mission.

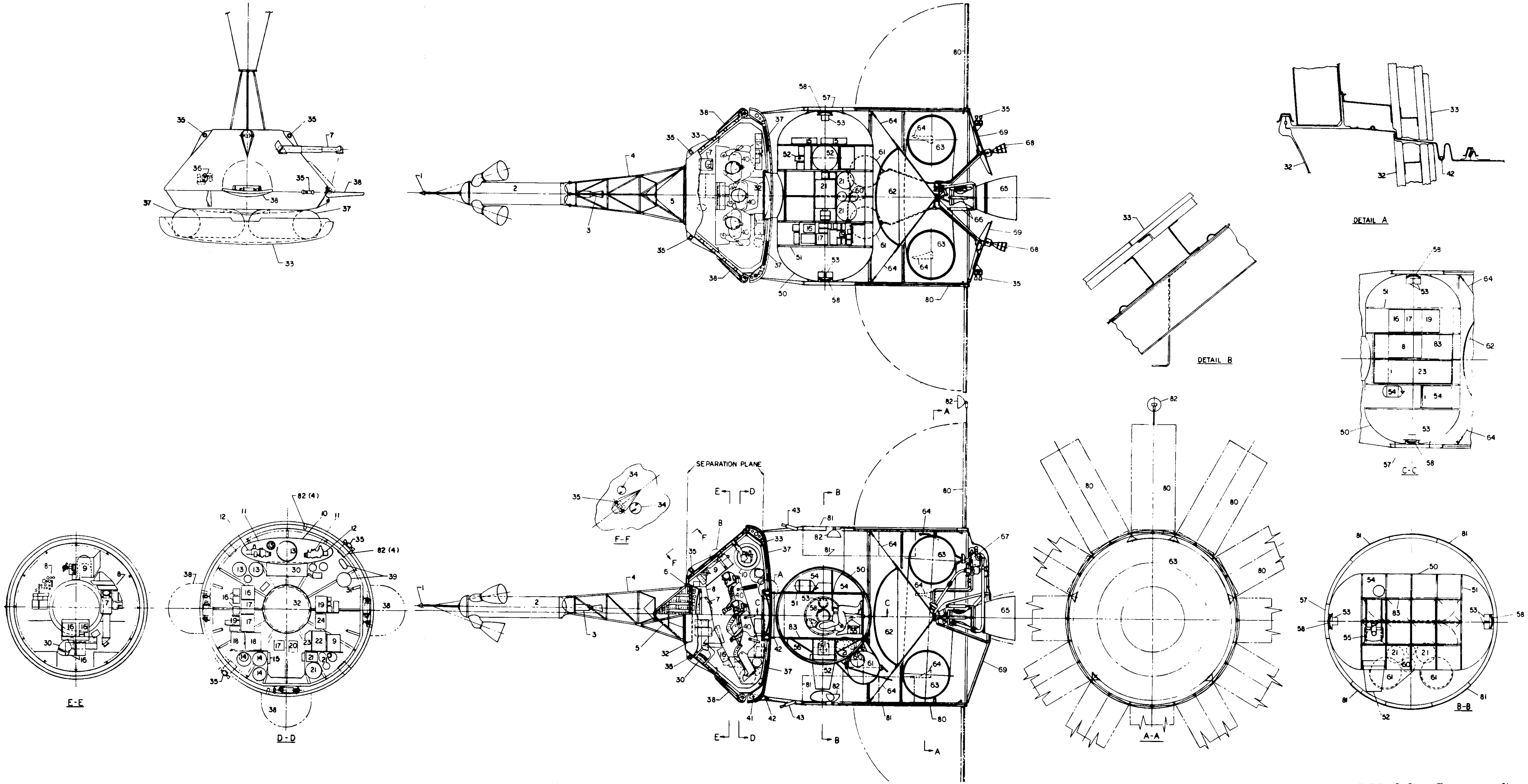
The arrangement of the aft structural shell, Mission Module and main tanks was developed simultaneously, in order that emphasis could be placed on an optimum combination;--as opposed to emphasizing the optimization of each item in itself.

[redacted]



W-1 Modular Spacecraft

1. PITOT HEAD
2. ESCAPE ROCKET
3. ESCAPE SYSTEM SEPARATION ROCKET
4. ESCAPE ROCKET SUPPORT TRUSS
5. PARACHUTE PACKAGE
6. PARACHUTE SEQUENCING EQUIPMENT
7. PERISCOPE AND DRIFT SIGHT
8. INSTRUMENT AND CONTROL PANEL
9. GUIDANCE EQUIPMENT
10. SAFETY COMPARTMENT
11. AUXILIARY POWER UNIT
12. ELECTRICAL GENERATOR
13. FUEL AND OXIDIZER TANKS FOR APU AND CONTROL SYSTEMS
14. O_2 AND N_2 SUPPLY TANKS
15. FILM STOWAGE
16. ELECTRICAL SYSTEM COMPONENTS
17. COMMUNICATION SYSTEM COMPONENTS
18. GUIDANCE AND NAVIGATION EQUIPMENT
19. TELEMETRY EQUIPMENT
20. RBE MEASURING EQUIPMENT
21. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (CABIN AND EQUIPMENT)
22. SURVIVAL KIT--3 MAN
23. WASTE DISPOSAL PACKAGE
24. DRINKING WATER SUPPLY
30. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (STRUCTURAL)
31. PRESSURE BULKHEAD
32. HATCH
33. HEAT SHIELD
34. WINDOW
35. REACTION CONTROL NOZZLES AND SERVOVALVES
36. SNORKEL
37. LANDING BAG
38. AERODYNAMIC CONTROL SURFACE AND ACTUATOR
39. GAS GENERATING EQUIPMENT
40. SEAT
41. HEAT SHIELD RELEASE MECHANISM
42. MODULE SEPARATION DEVICE
43. MODULE SEPARATION ROCKETS
50. MISSION MODULE
51. EQUIPMENT SUPPORTING STRUCTURE
52. CAMERA
53. SCIENTIFIC DATA GATHERING EQUIPMENT
54. FOOD AND WATER STOWAGE
55. WASTE COLLECTION
56. WATER RECOVERY UNIT
57. MAIN ENTRANCE HATCH
58. VIEWING PORT
60. H PRESSURE BOTTLE
61. FUEL AND OXIDIZER TANKS--REACTION CONTROL AND VERNIER ROCKET SYSTEM
62. MAIN LOX TANK
63. MAIN LH TANK
64. TANK SUPPORT AND INSULATING STRUCTURE
65. MAIN ENGINE (XLR-119 MODIFIED)
66. GIMBAL ACTUATORS
67. PUMP
68. VERNIER ROCKET NOZZLE
69. FLAME SHIELD
80. SOLAR CELL ARRAY
81. SPACE RADIATOR
82. ANTENNA
83. HINGED RADIATION SHIELD



L2C Modular Spacecraft

1. PITOT HEAD
2. ESCAPE ROCKET
3. ESCAPE SYSTEM SEPARATION ROCKET
4. ESCAPE ROCKET SUPPORT TRUSS
5. PARACHUTE AND ROCKET PACKAGE
6. PARACHUTE SEQUENCING EQUIPMENT
7. PERISCOPE AND DRIFT SIGHT
8. INSTRUMENT AND CONTROL PANEL
9. GUIDANCE EQUIPMENT
10. SAFETY COMPARTMENT
11. AUXILIARY POWER UNIT
12. ELECTRICAL GENERATOR
13. FUEL AND OXIDIZER TANKS FOR APU AND CONTROL SYSTEMS
14. O_2 AND N_2 SUPPLY TANKS
15. FILM STORAGE
16. ELECTRICAL SYSTEM COMPONENTS
17. COMMUNICATION SYSTEM COMPONENTS
18. GUIDANCE AND NAVIGATION EQUIPMENT
19. TELEMETRY EQUIPMENT
20. RDE MEASURING EQUIPMENT
21. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (CABIN AND EQUIPMENT)
22. SURVIVAL KIT

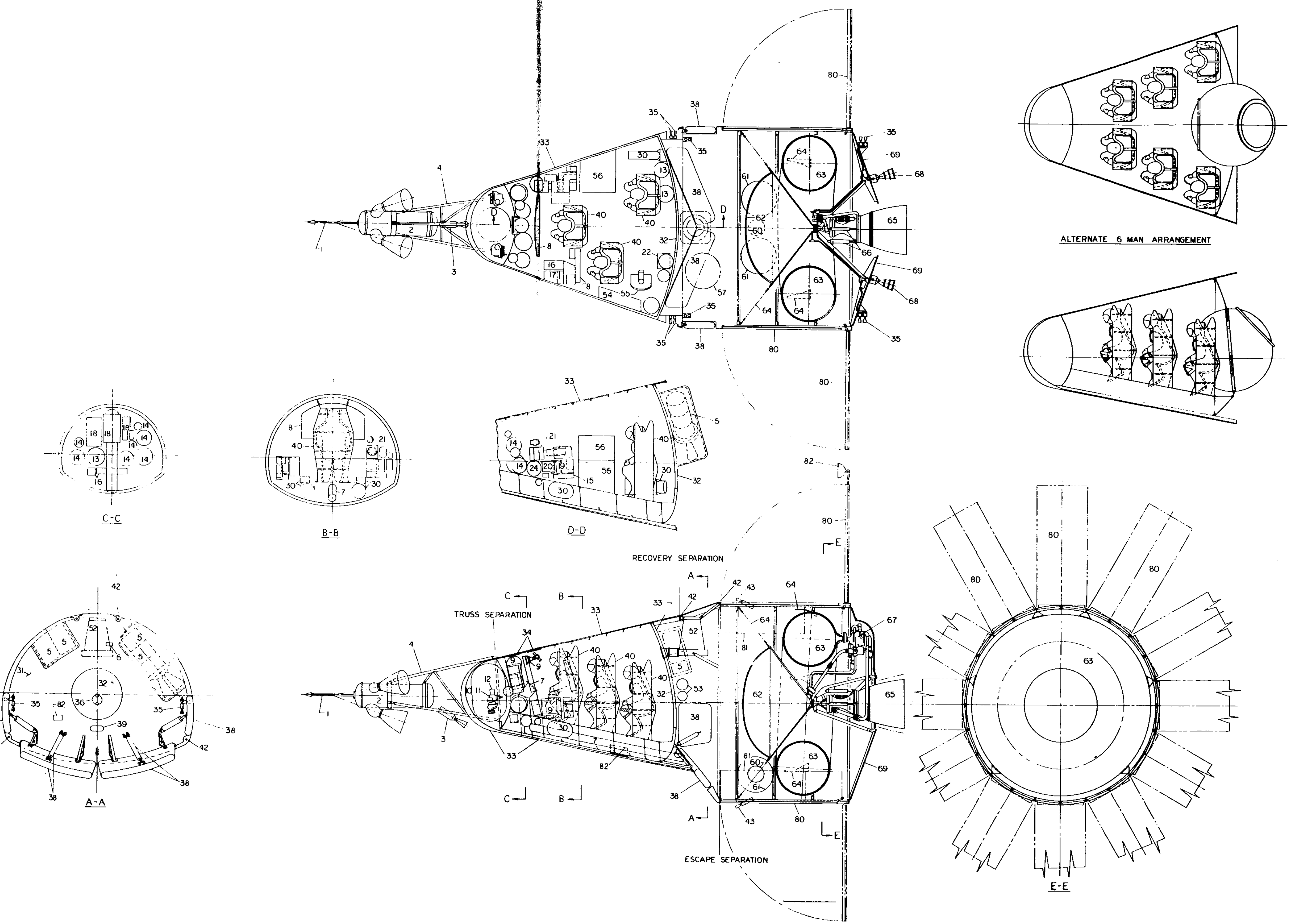
24. DRINKING WATER SUPPLY
30. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (STRUCTURAL)
31. PRESSURE BULKHEAD
32. HATCH
33. HEAT SHIELD
34. WINDOW
35. REACTION CONTROL NOZZLES AND SERVOVALVES
36. SNORKEL

38. AERODYNAMIC CONTROL SURFACE AND ACTUATOR
39. GAS GENERATING EQUIPMENT
40. SEAT

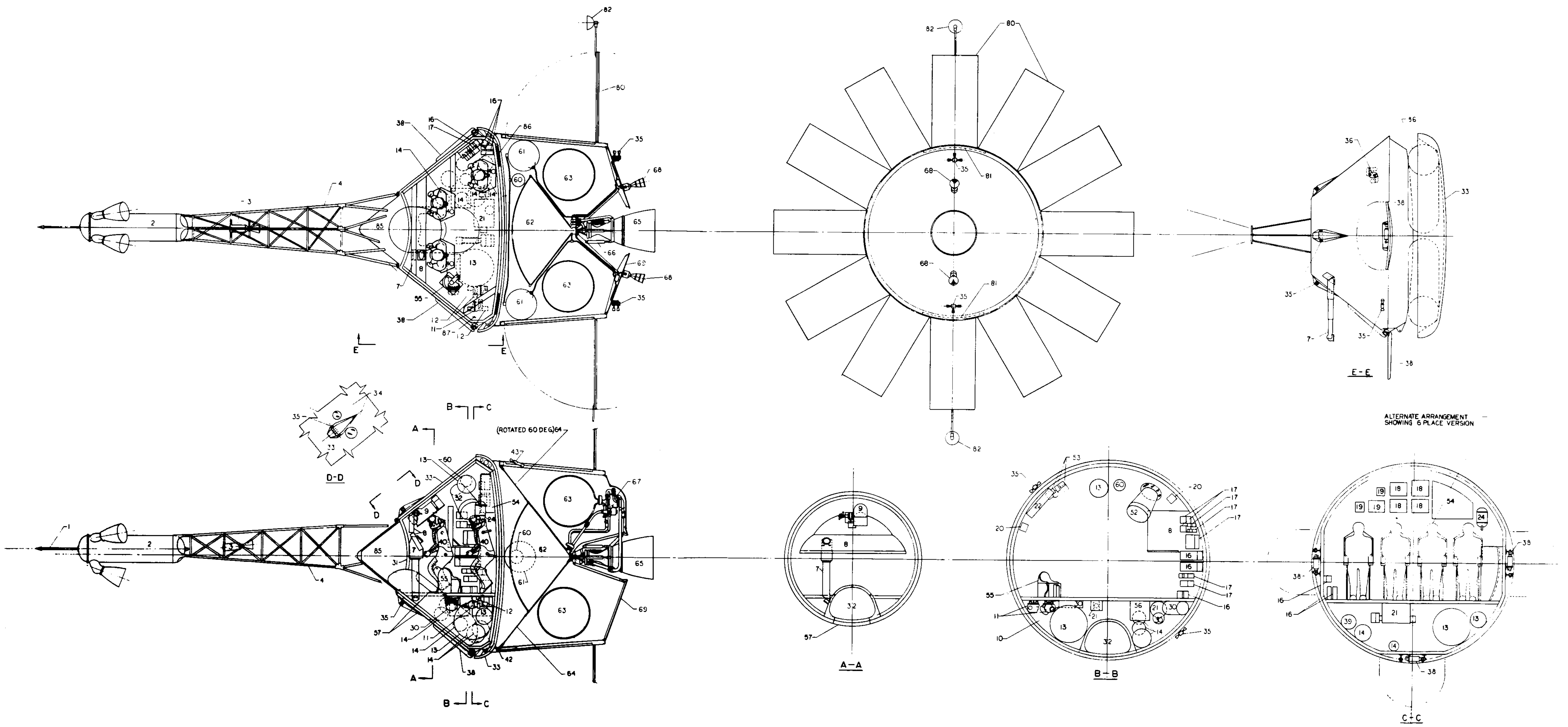
42. MODULE SEPARATION DEVICE
43. MODULE SEPARATION ROCKETS

52. CAMERA
53. SCIENTIFIC DATA GATHERING EQUIPMENT
54. FOOD AND WATER STORAGE
55. WASTE COLLECTION
56. WATER RECOVERY UNIT
57. MAIN ENTRANCE HATCH

60. H_e PRESSURE BOTTLE
61. FUEL AND OXIDIZER TANKS—REACTION CONTROL AND VERNIER ROCKET SYSTEM
62. MAIN LOX TANK
63. MAIN LH TANK
64. TANK SUPPORT AND INSULATING STRUCTURE
65. MAIN ENGINE DXLR-119 MODIFIED
66. GIMBAL ACTUATORS
67. PUMP
68. VERNIER ROCKET NOZZLE
69. FLAME SHIELD
80. SOLAR CELL ARRAY
81. SPACE RADIATOR
82. ANTENNA

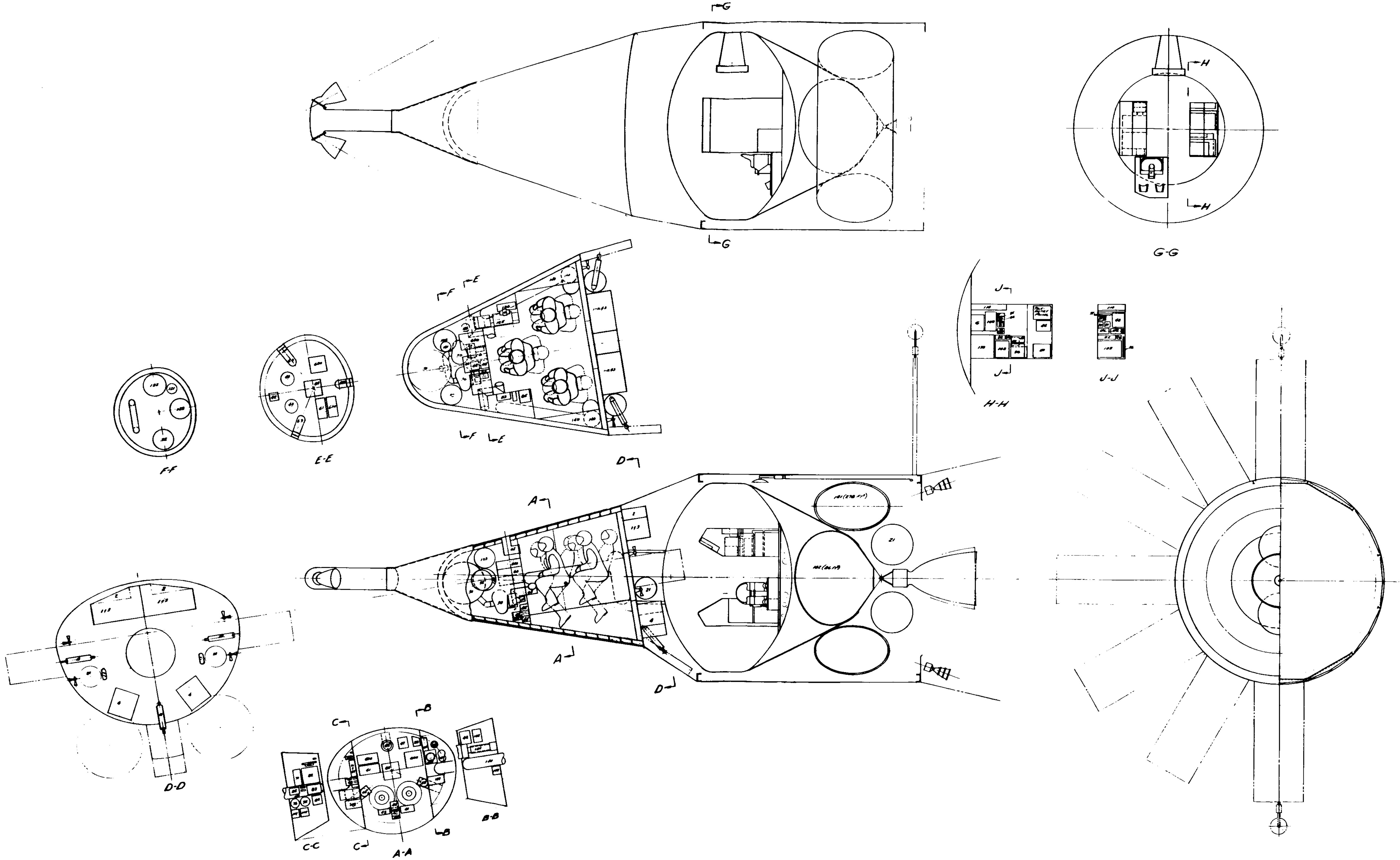


W-1 Integrated Spacecraft

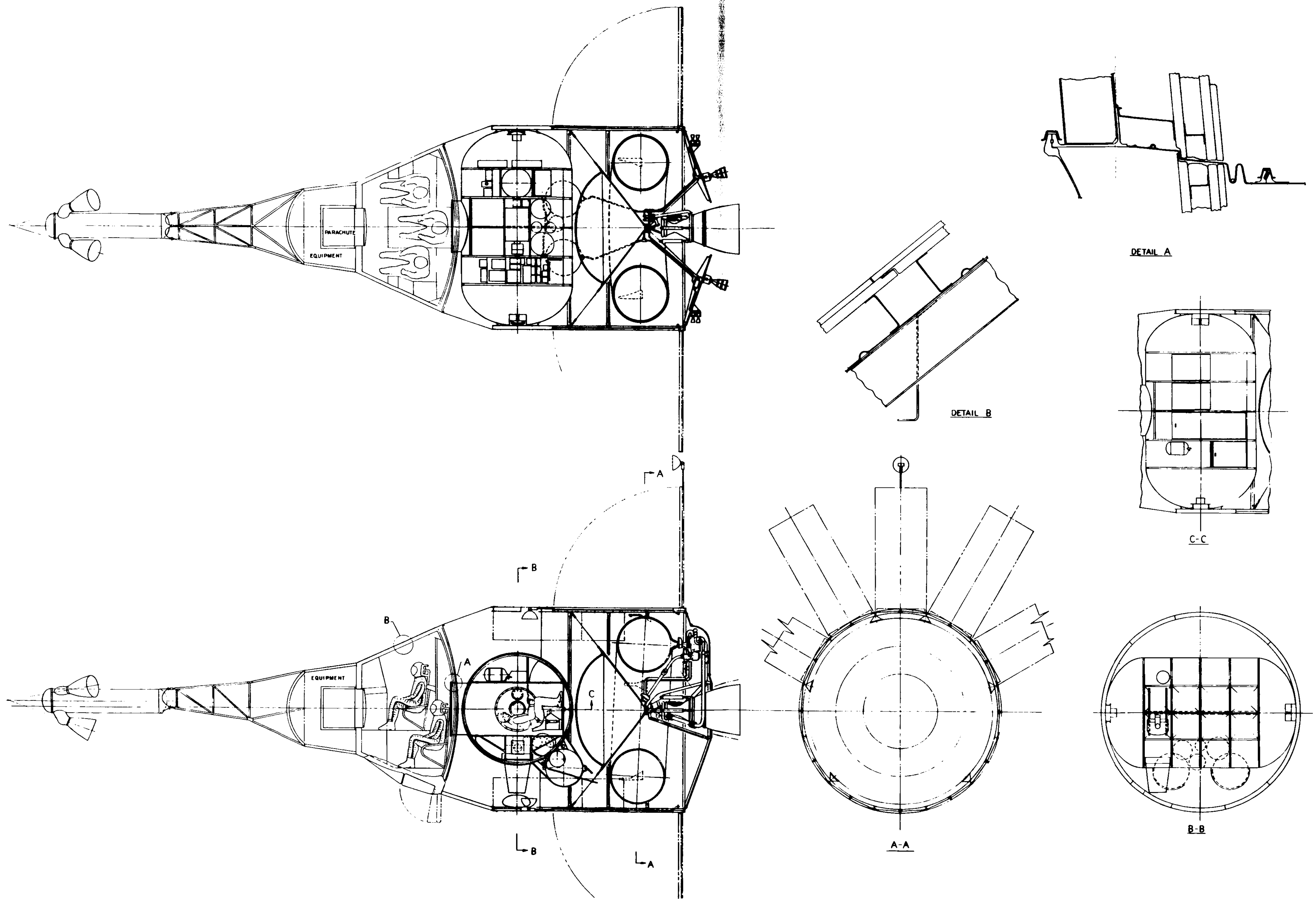


1. PITOT HEAD
2. ESCAPE ROCKET
3. ESCAPE SYSTEM SEPARATION ROCKET
4. ESCAPE ROCKET SUPPORT TRUSS
7. PERISCOPE AND DRIFT SIGHT
8. INSTRUMENT AND CONTROL PANEL
9. GUIDANCE EQUIPMENT
10. SAFETY COMPARTMENT
11. AUXILIARY POWER UNIT
12. ELECTRICAL GENERATOR
13. FUEL AND OXIDIZER TANKS FOR APU AND CONTROL SYSTEMS
14. O_2 AND N_2 SUPPLY TANKS
16. ELECTRICAL SYSTEM COMPONENTS
17. COMMUNICATION SYSTEM COMPONENTS
18. GUIDANCE AND NAVIGATION EQUIPMENT
19. TELEMETRY EQUIPMENT
20. RBE MEASURING EQUIPMENT
21. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (CABIN AND EQUIPMENT)
22. SURVIVAL KIT
24. DRINKING WATER SUPPLY
30. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (STRUCTURAL)
31. PRESSURE BULKHEAD
32. AIR LOCK
33. HEAT SHIELD
34. WINDOW
35. REACTION CONTROL NOZZLES AND SERVOVALVES
36. SNORKEL
38. AERODYNAMIC CONTROL SURFACE AND ACTUATOR
39. GAS GENERATING EQUIPMENT
40. SEAT
42. MODULE SEPARATION DEVICE
43. MODULE SEPARATION ROCKETS
52. CAMERA
53. SCIENTIFIC DATA GATHERING EQUIPMENT
54. FOOD AND WATER STORAGE
55. WASTE COLLECTION
56. WATER RECOVERY UNIT
57. MAIN ENTRANCE HATCH
60. H_2 PRESSURE BOTTLE
61. FUEL AND OXIDIZER TANKS--REACTION CONTROL AND VERNIER ROCKET SYSTEM
62. MAIN LOX TANK
63. MAIN LH TANK
64. TANK SUPPORT AND INSULATING STRUCTURE
65. MAIN ENGINE (XLR-119 MODIFIED)
66. GIMBAL ACTUATORS
67. PUMP
68. VERNIER ROCKET NOZZLE
69. FLAME SHIELD
80. SOLAR CELL ARRAY
81. SPACE RADIATOR
82. ANTENNA
85. PARACHUTE PACK AND SEQUENCING EQUIPMENT
86. LANDING BAG
87. WASTE DISPOSAL

L2C Integrated Spacecraft

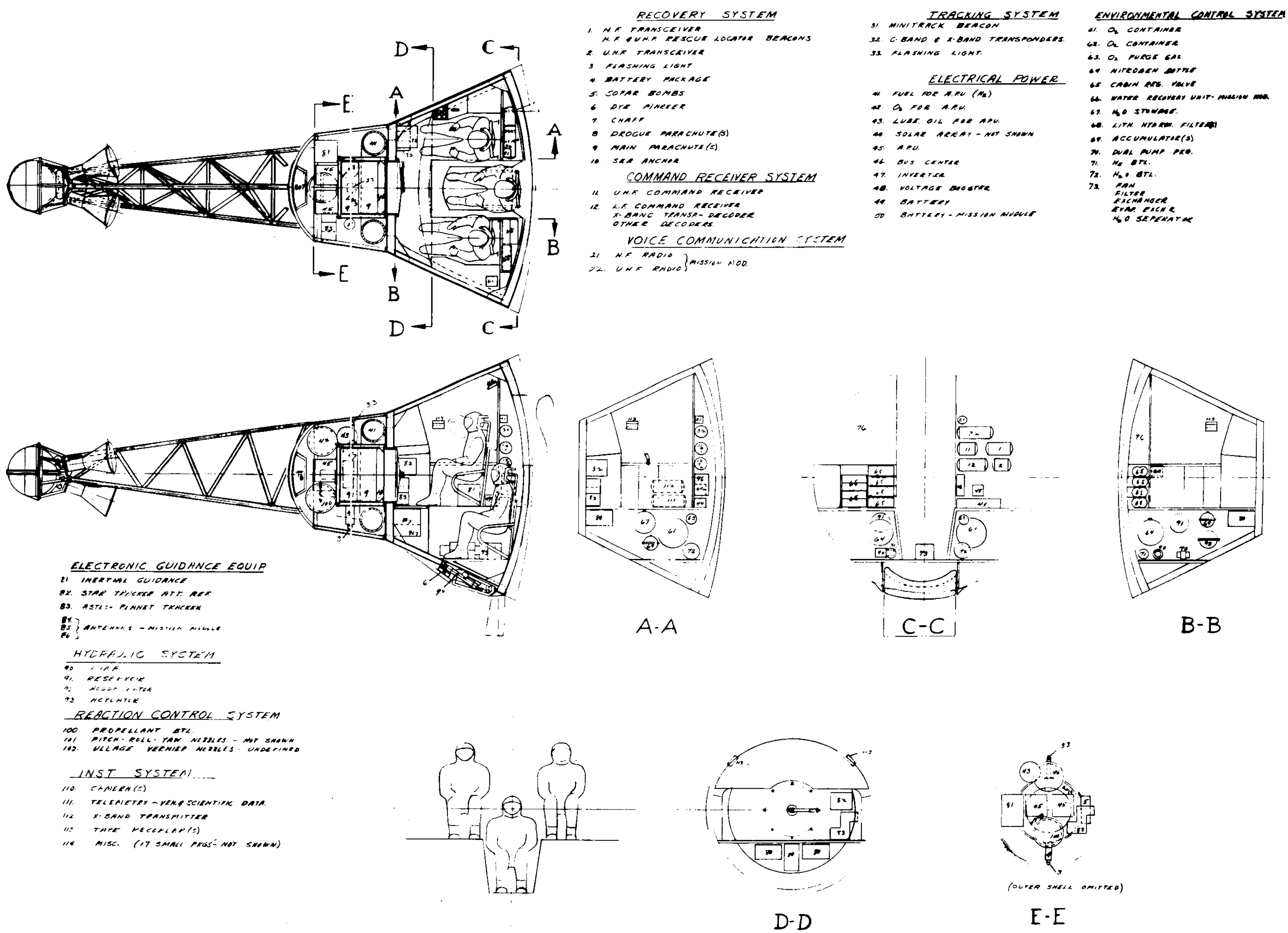


M-1-1 Modular Spacecraft



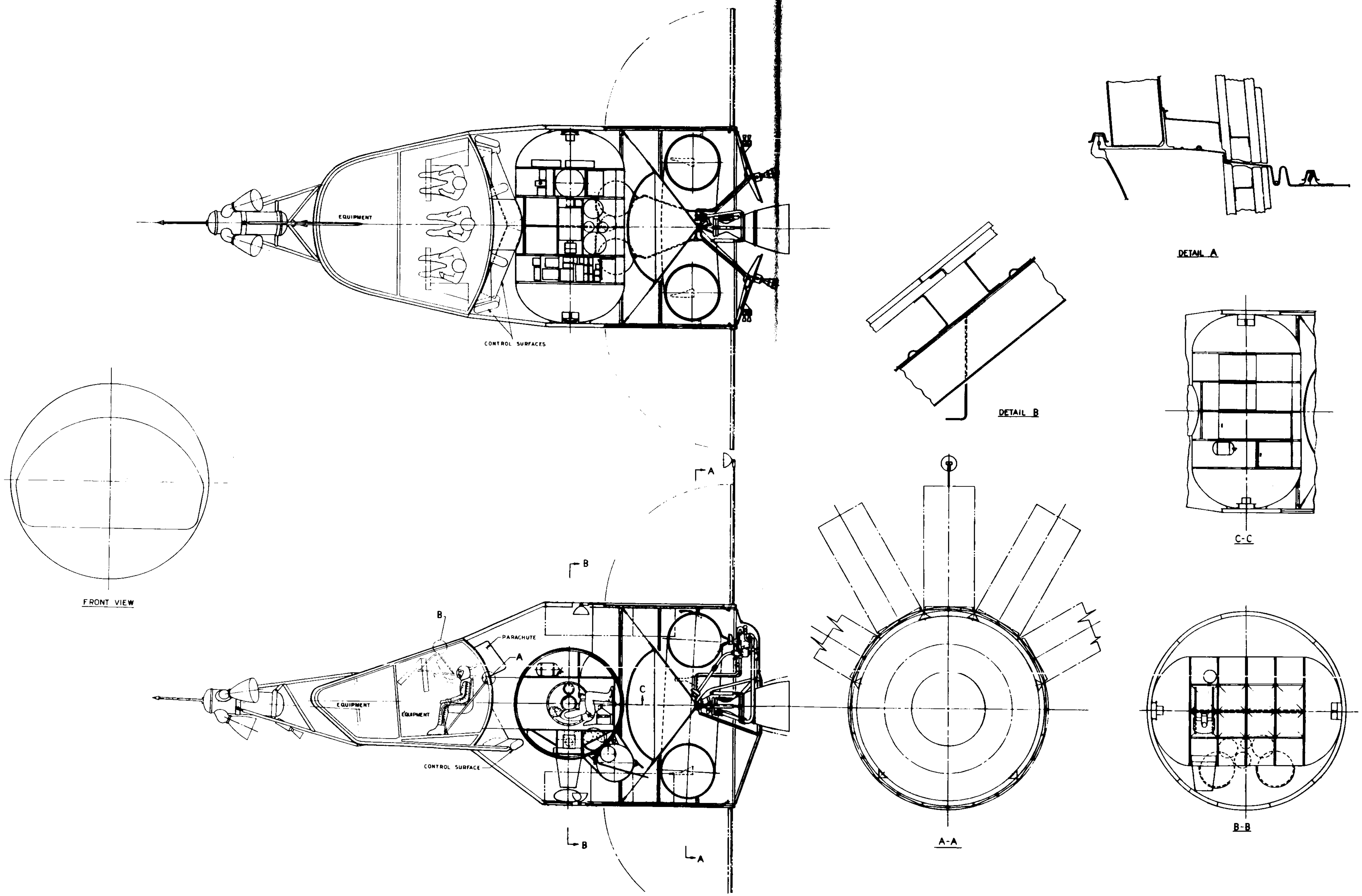
Flapped Mercury Modular Spacecraft

[redacted]

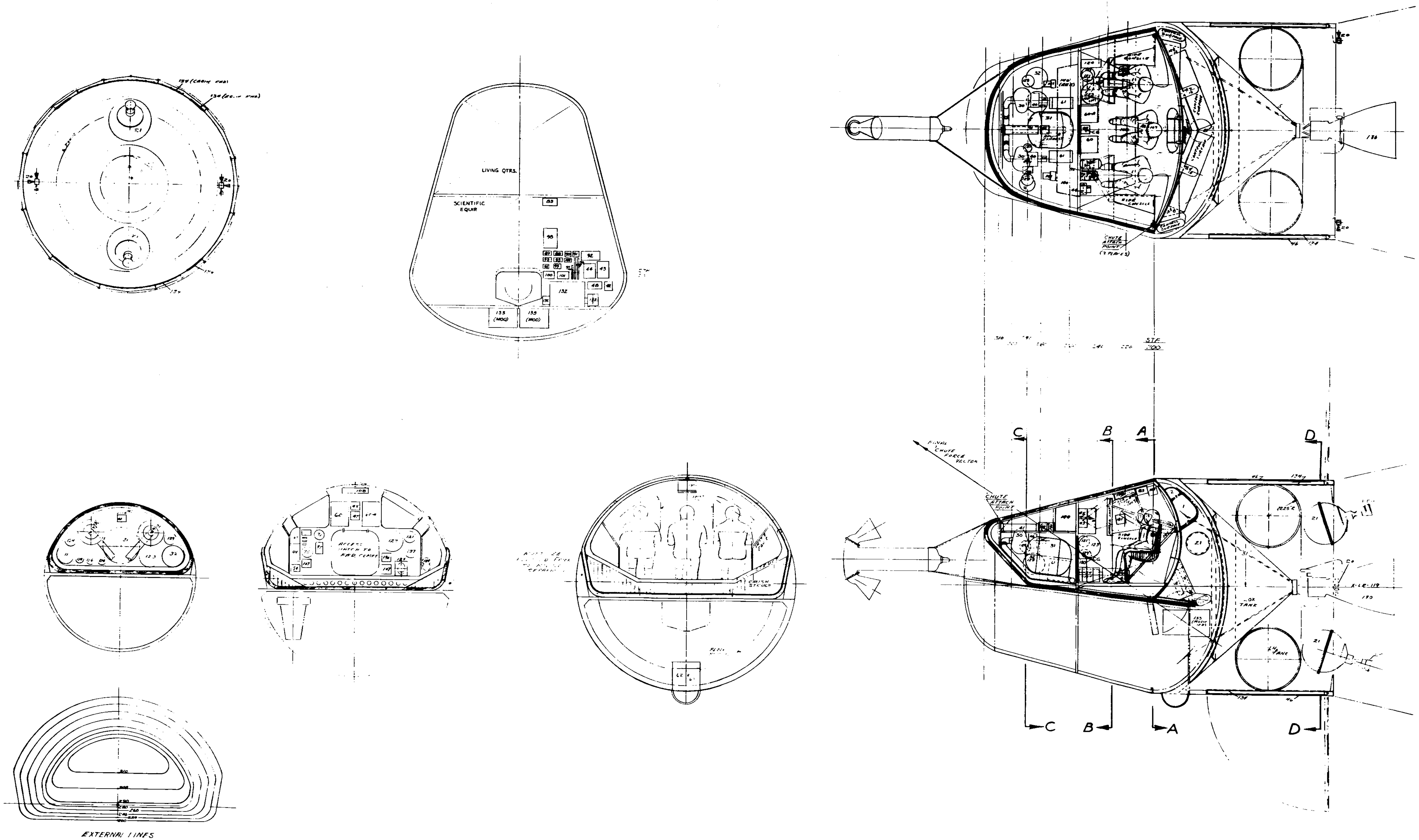


Flapped Mercury Re-entry Vehicle

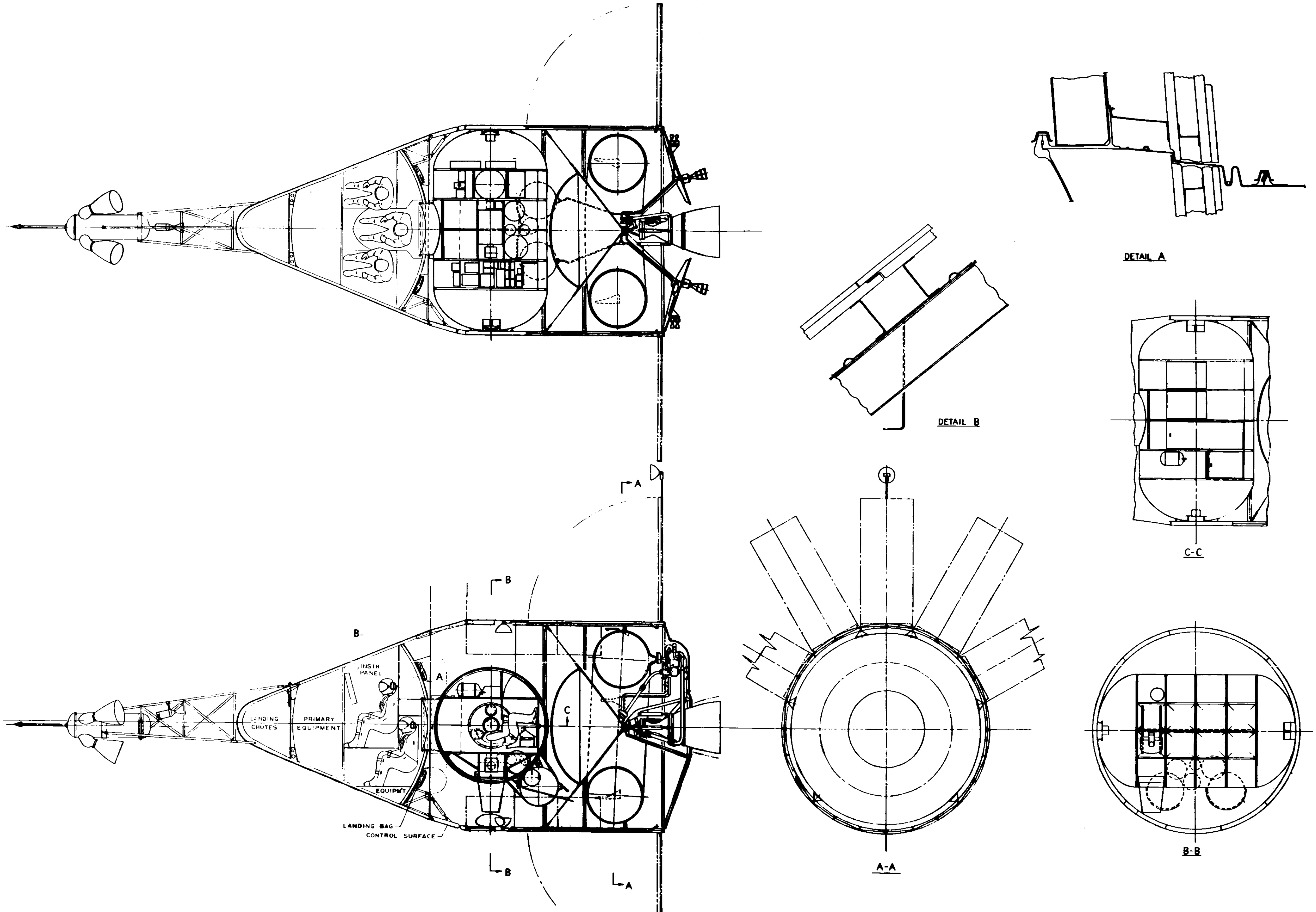
[redacted]



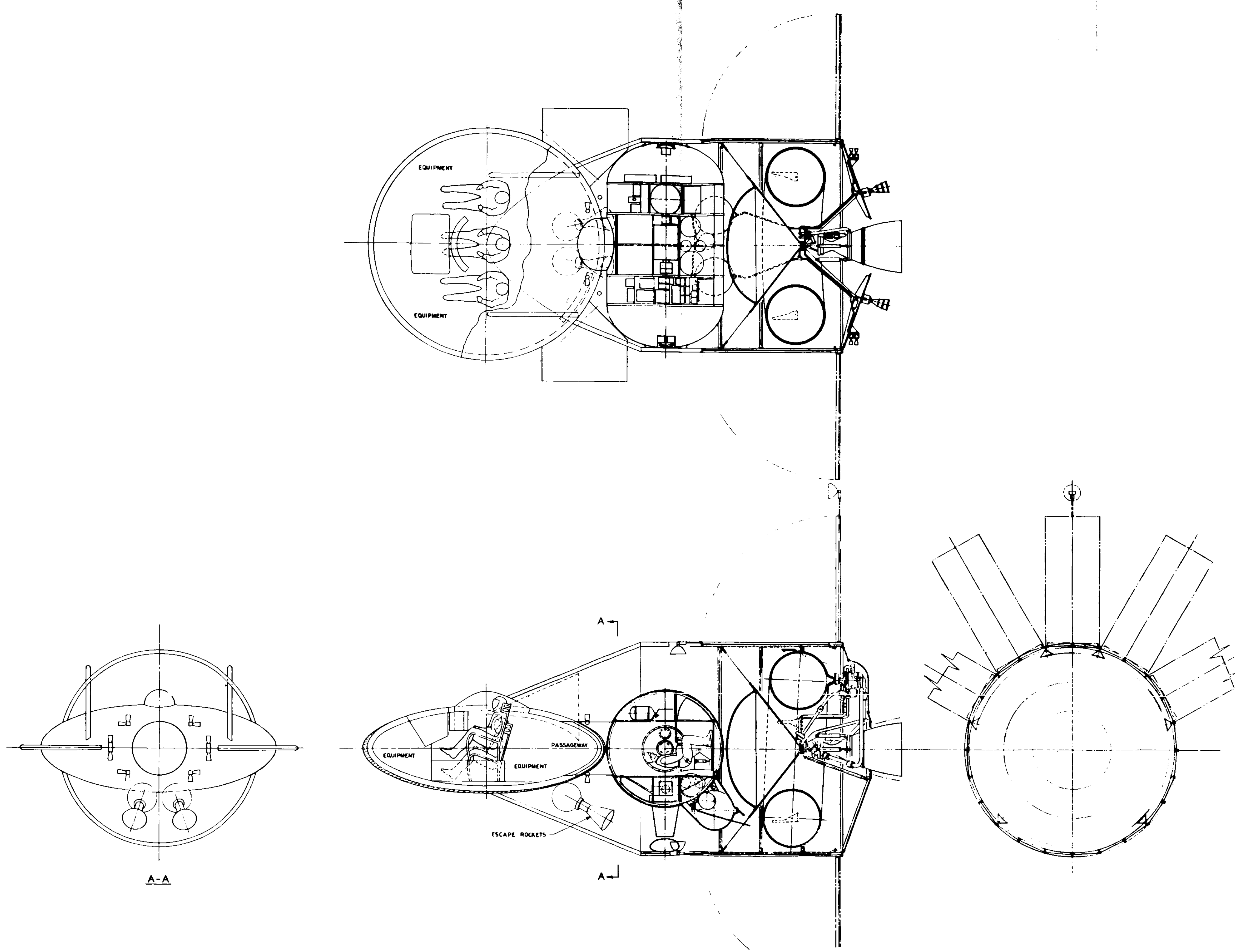
L-1 Modular Spacecraft



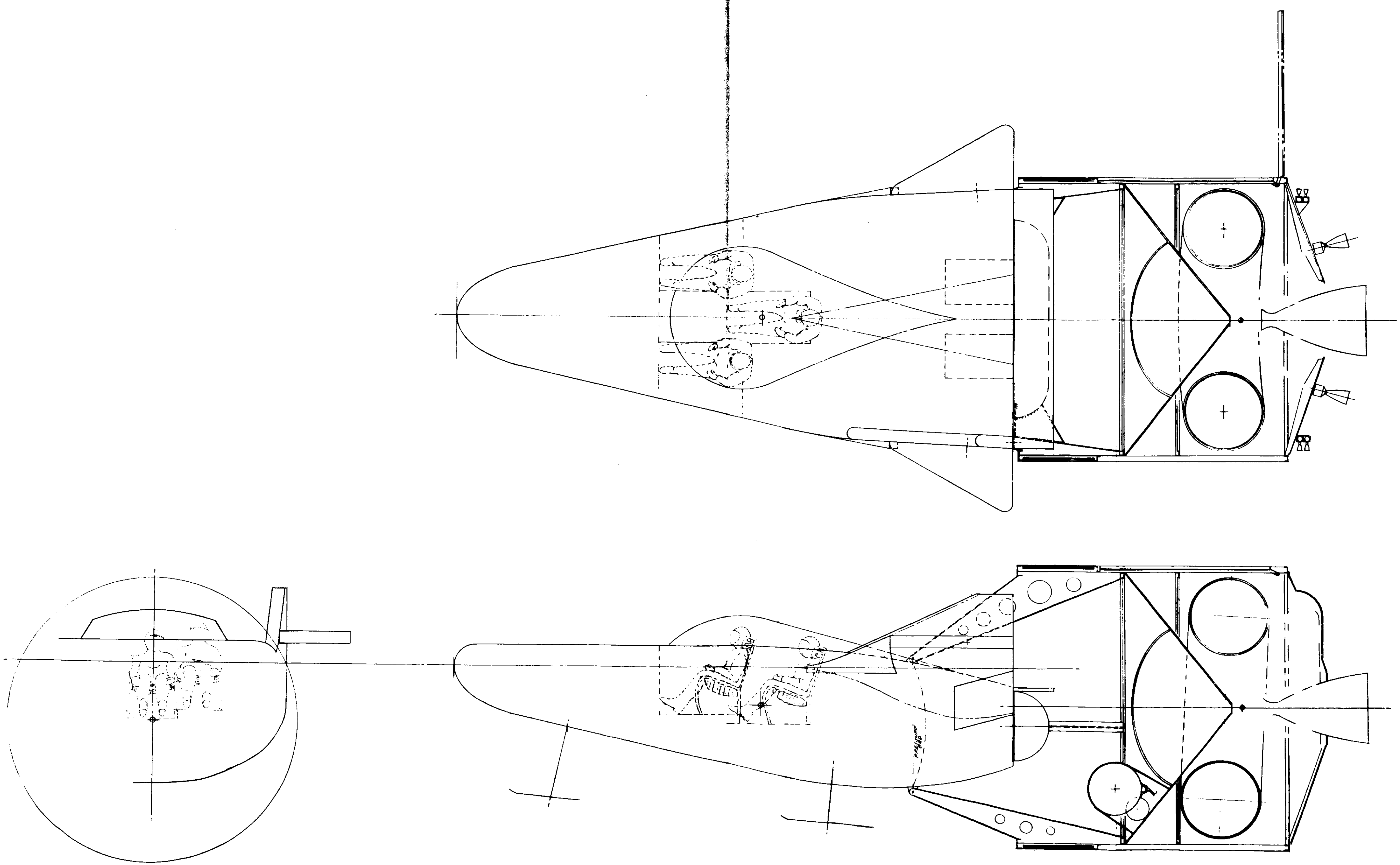
L-1 Modular Spacecraft (side by side mission module)



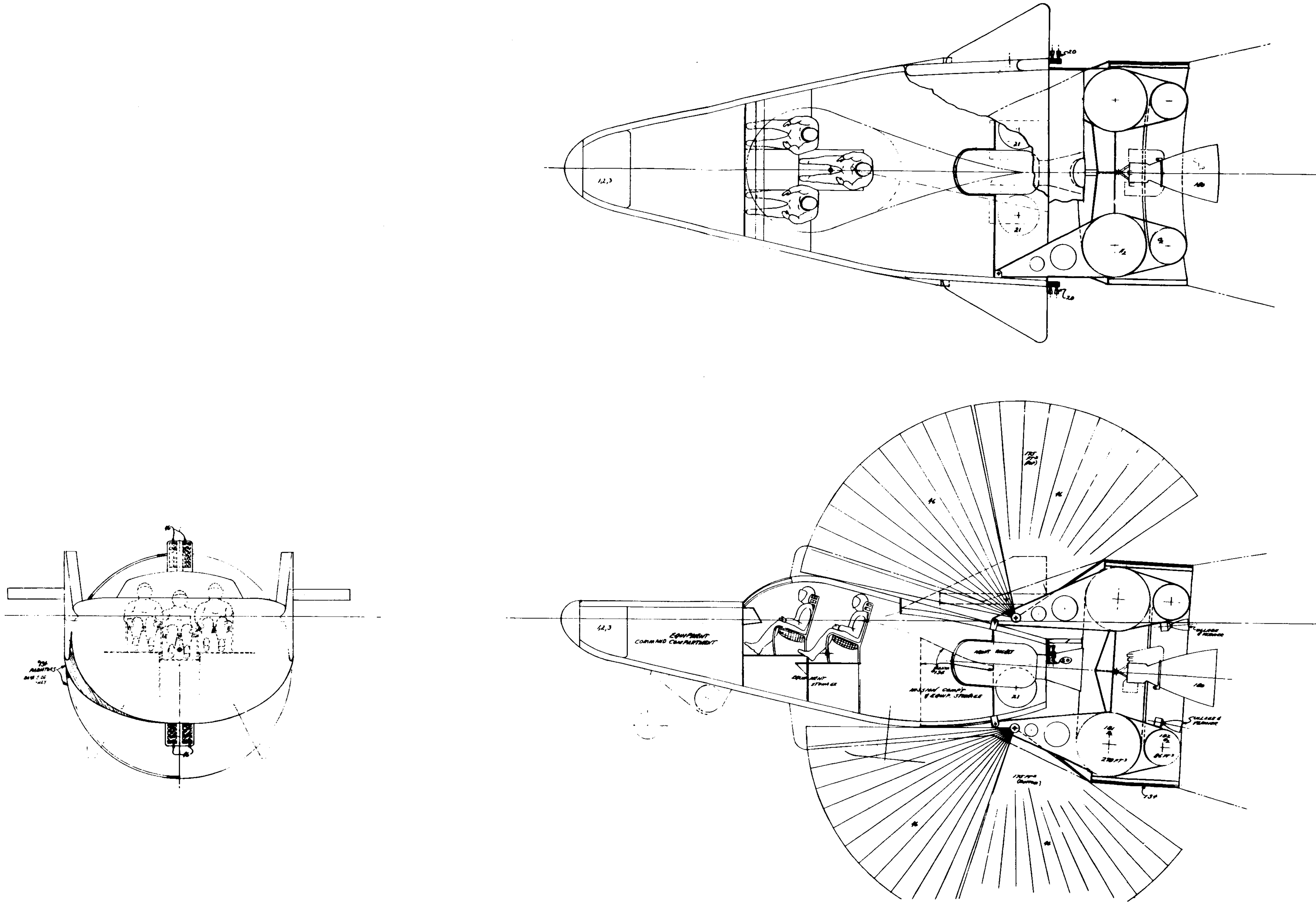
L-8 Modular Spacecraft



Lenticular Modular Spacecraft



M-2B Semiintegrated Spacecraft



M-2B Integrated Spacecraft

Tracking & Communications

TM-25

APOLLO MID-TERM HANDOUT

TECHNICAL MEMORANDUM NO. 25

TRACKING COMMUNICATIONS AND INSTRUMENTATION

10 March 1961

J. Pipkin

~~CONFIDENTIAL~~

Apollo Technical Memorandum -25

March 14, 1961

<u>Tracking, Communication and Instrumentation</u>

A brief collection of current working data reflecting the principal characteristics of the system is presented based on a number of established guide lines. The salient information has been briefly summarized in a group of curves, charts, tables, and block diagrams.

Tracking coverage for missions representing both a maximum northern and southern lunar declination as well as a zero declination have been included and the functional block diagrams of the various on-board equipments given. Since the complete lunar mission involves a number of phases, different equipments and combinations of these equipments will be utilized.

Although studies are still continuing, certain approaches to the problem reflect the compromises brought about by a number of constraints. For instance, decentralization in tracking and trajectory determination will be employed as much as possible during the early program due to the limited availability of reliable long haul channels. However, as more dependable world-wide communication nets become available, centralized control will be maintained. In any event, full use of established ground facilities with as few additions to the existing complex as possible will be employed to permit

a) Launch at AMR, deep space monitoring by DSIF, re-entry at PMR, and landing at Edwards AFB.

b) Vehicle monitoring during the various mission phases.

c) Assistance to the on-board guidance system by providing necessary data inputs for use in up-dating the on-board equipment.

~~CONFIDENTIAL~~

Communications will involve two categories which consist of the information transfer from the ground complexes to the vehicle, and the information transfer among the various elements of the ground complexes themselves. The overall backup reliability of the ground system will not degrade the task of safe vehicle return. The electronic communications and tracking circuits of the on-board equipment, while employing advanced transistorized techniques, will be amenable to the use of grown functional units as they become operationally available.

On-board instrumentation will, in general, cover four significant areas:

a) Vehicle and vehicle systems

b) Life support and human reaction to environment

c) Scientific measurements for crew safety

d) Lunar surveillance

Finally, the data processing system for instrumentation will consist of two types: one compatible with the up-dated Mercury ground system and used for <u>injection</u>, <u>earth orbit</u> and <u>re-entry</u> phases; and, the other one compatible with the anticipated DSIF and used for the <u>deep space</u> phases.

[redacted]

[illegible]
[illegible]
[illegible]
[illegible]
[illegible] Station
[illegible] Communications Complex
Launch and Injection Tracking **(Omitted)**
Deep Space Tracking Coverage
Re-entry Tracking Southerly Declination
Re-entry Tracking 0° Declination
0° Declination Trajectory Re-entry to Touchdown Coverage
Re-entry Tracking Northerly Declination
Northerly Declination Re-entry to Touchdown Coverage
Earth Orbital Mission Coverage
Tracking Stations Descriptions
APOLLO Computational Facilities

Vehicle System

	Functions
General Vehicle System	Communications Transponders, Data Processing, and Instrumentation

[illegible]ions - Transponders (Aided Tracking and Rescue)

VHF Transmitter (230 mc.)	Range T/M and communications
VHF Receiver (230 mc.)	Range T/M and communications
S Band Transmitter	[illegible] T/M and communications
S Band Transmitter-Receiver Phase Lock Loop	
S Band Receiver	[illegible] T/M and communications
VHF Transponder (132 mc.)	Minitrack Response and Communications
UHF Transmitter-Receiver (300 mc.)	Rescue Communications Beacon
H.F. Transmitter Receiver	Rescue Communications Beacon
Re-[illegible] Transmitter Beacon	Re-entry Communications
C and X Band Beacons	C Band - FPS/16 SPS 56 Tracking Pre and Post re-entry; X Band - Boost Tracking

[redacted]

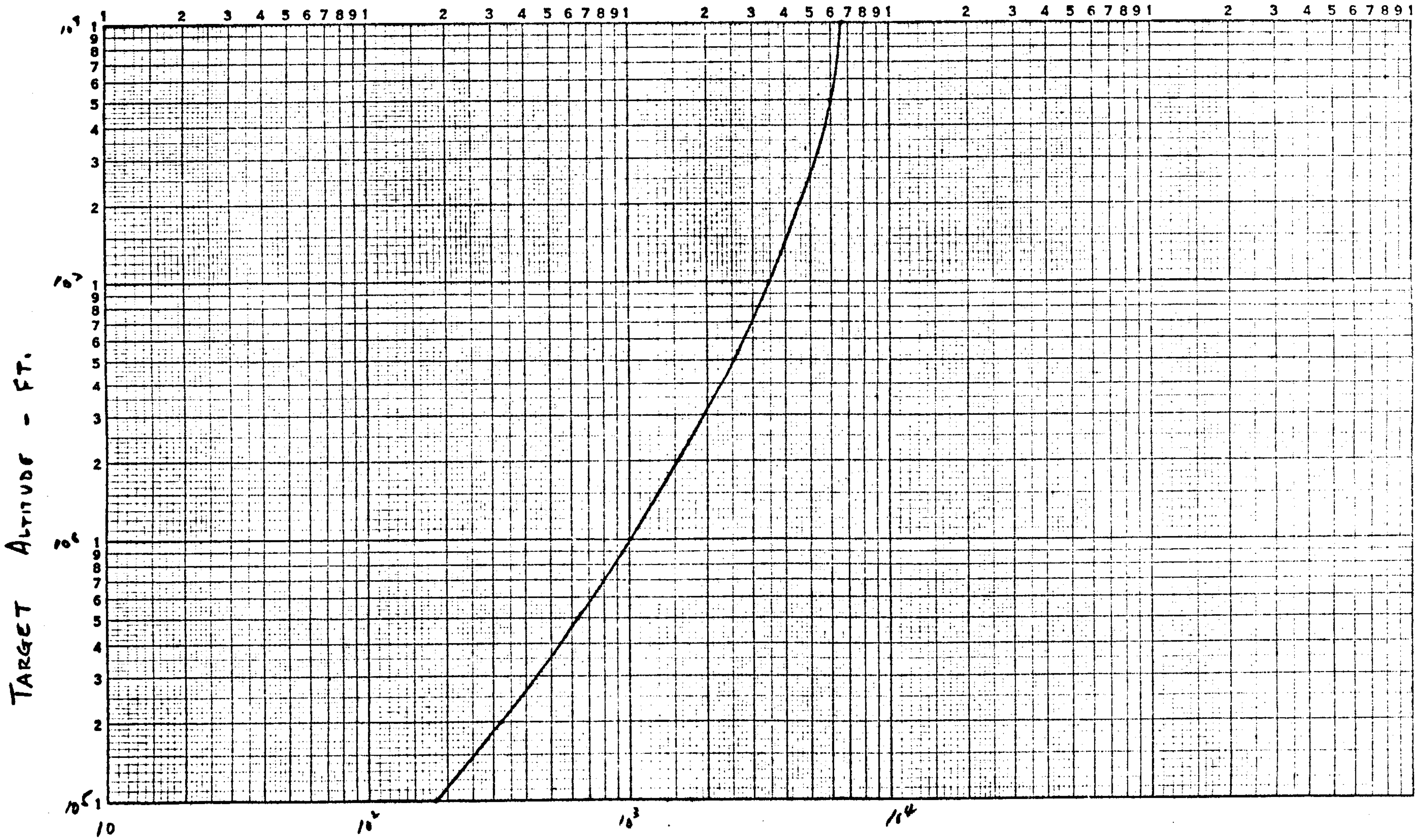
Table of Contents (cont'd)

Figure No.	Title	Function
TM25-25	APOLLO Vehicle Reception System	Demodulator-Decoder
TM25-26	Vehicle Reception Signal Processing	Receiver Signal Processing-Signal Selector (typical)

<u>Instrumentation:</u>

Figure No.	Title	Function
TM25-27	Pulse Code Modulation System	Deep Space Digital System provides on board data processing of slowly varying analog and digital telemetry signals
TM25-28A 28B 28C	Typical Instrumentation Measurement Subsystems	
TM25-29	Data Translation and Recording System	Provides for on board recording of communications information and data translation system for stacking of PCM, Voice and PDM/FM for use on single transmitter
TM25-30	Malfunction Detection System	Provides capability of on board evaluation of vehicle system performance.

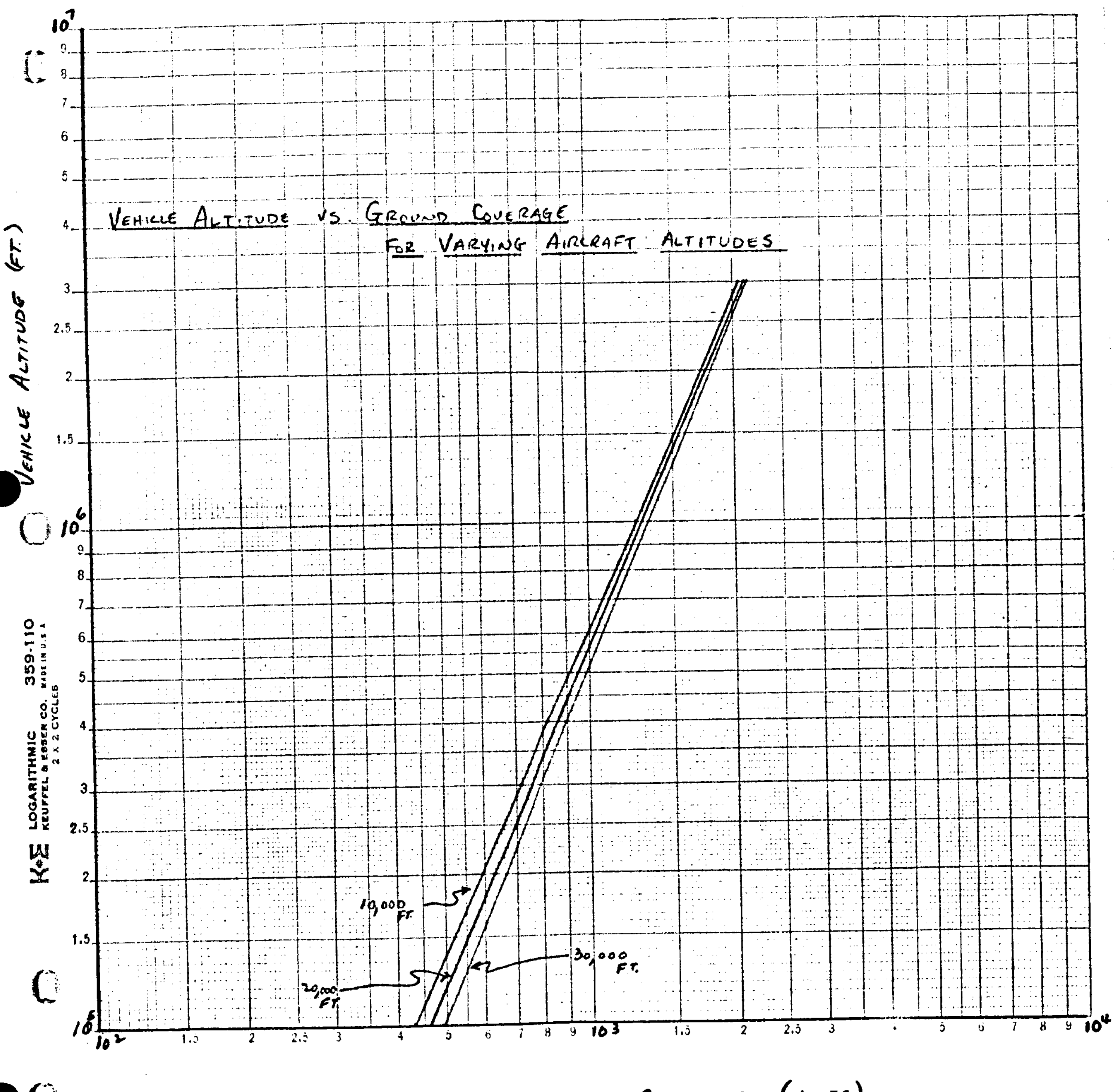
GROUND RADAR COVERAGE - 5° TERRAIN MASK - $\frac{4}{3}$ EARTH RADIUS



TM 25-1

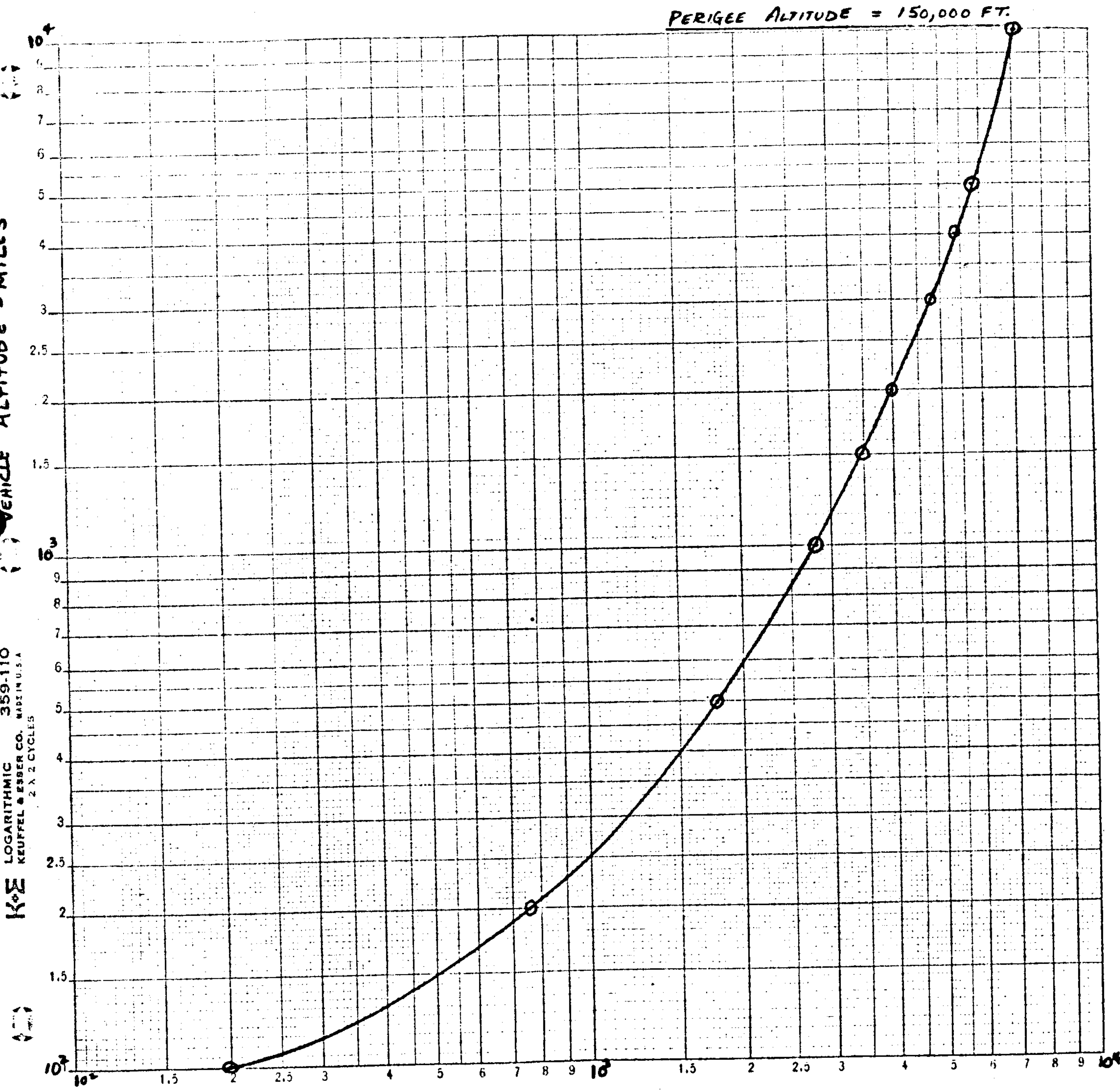
AIRBORNE RADAR COVERAGE

RADAR DEPRESSED TO 1° OF HORIZON

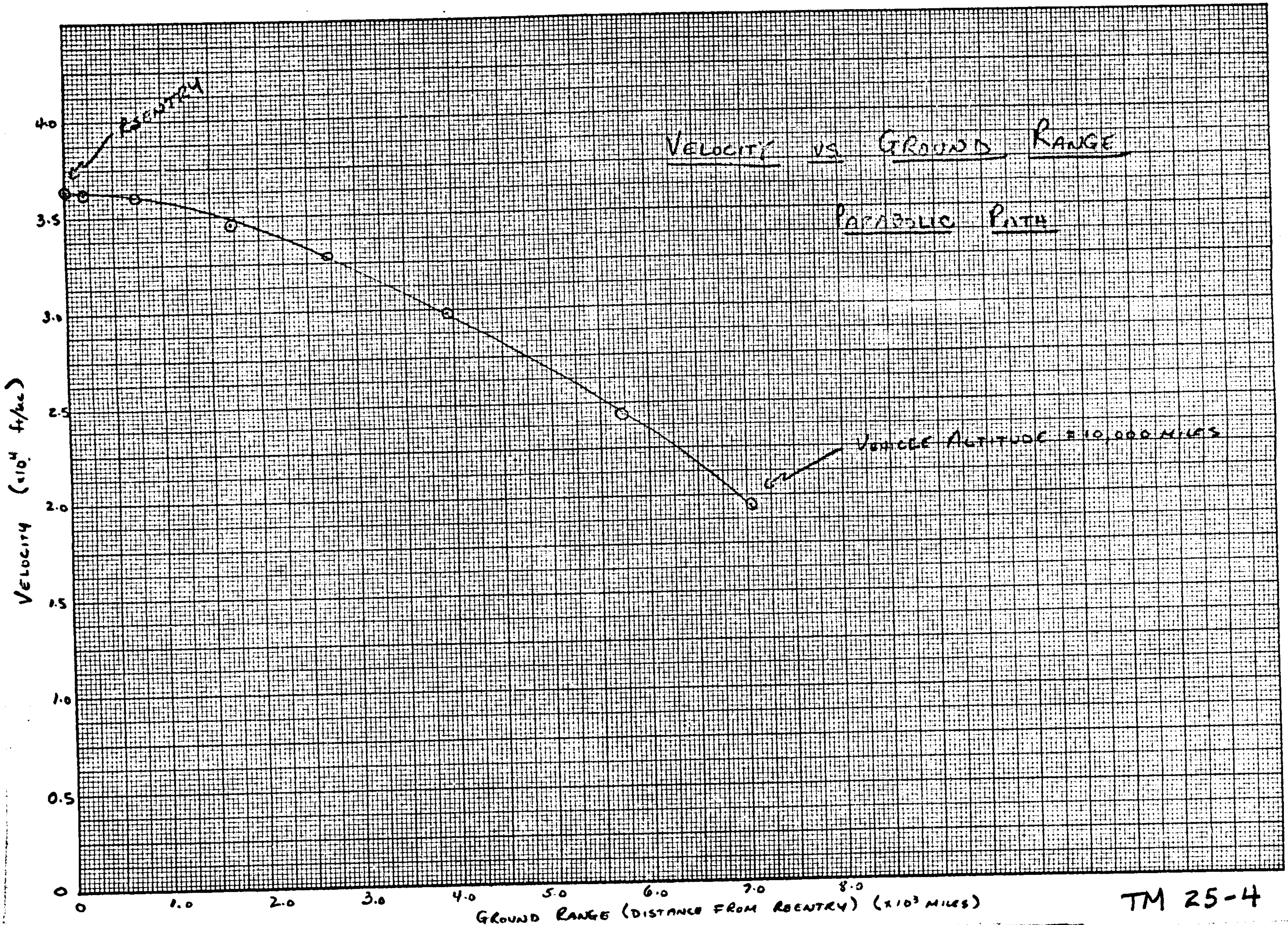


ALTITUDE VS. GROUND RANGE

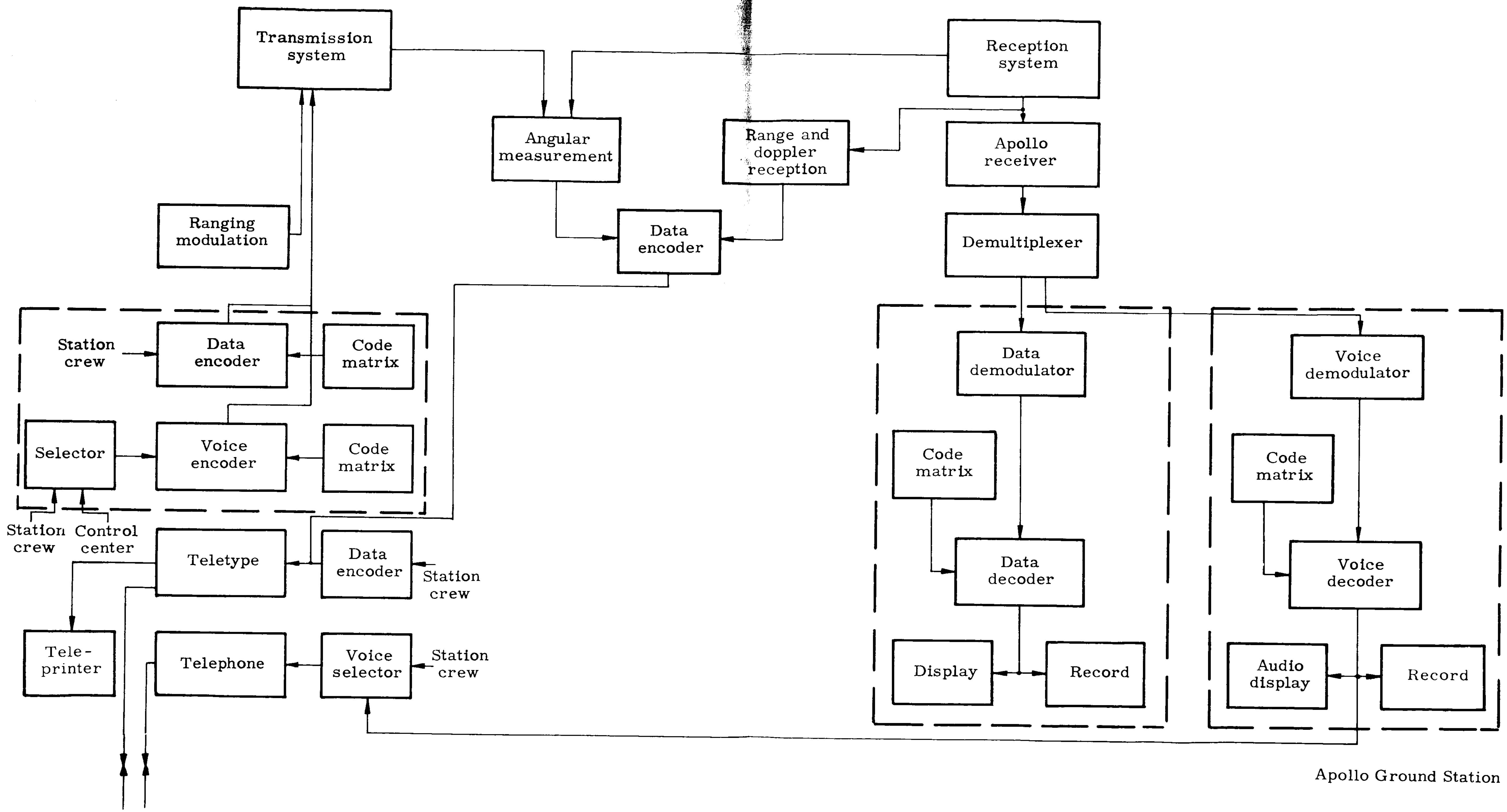
PARABOLIC PATH   10,000 MILE ALTITUDE TO REENTRY



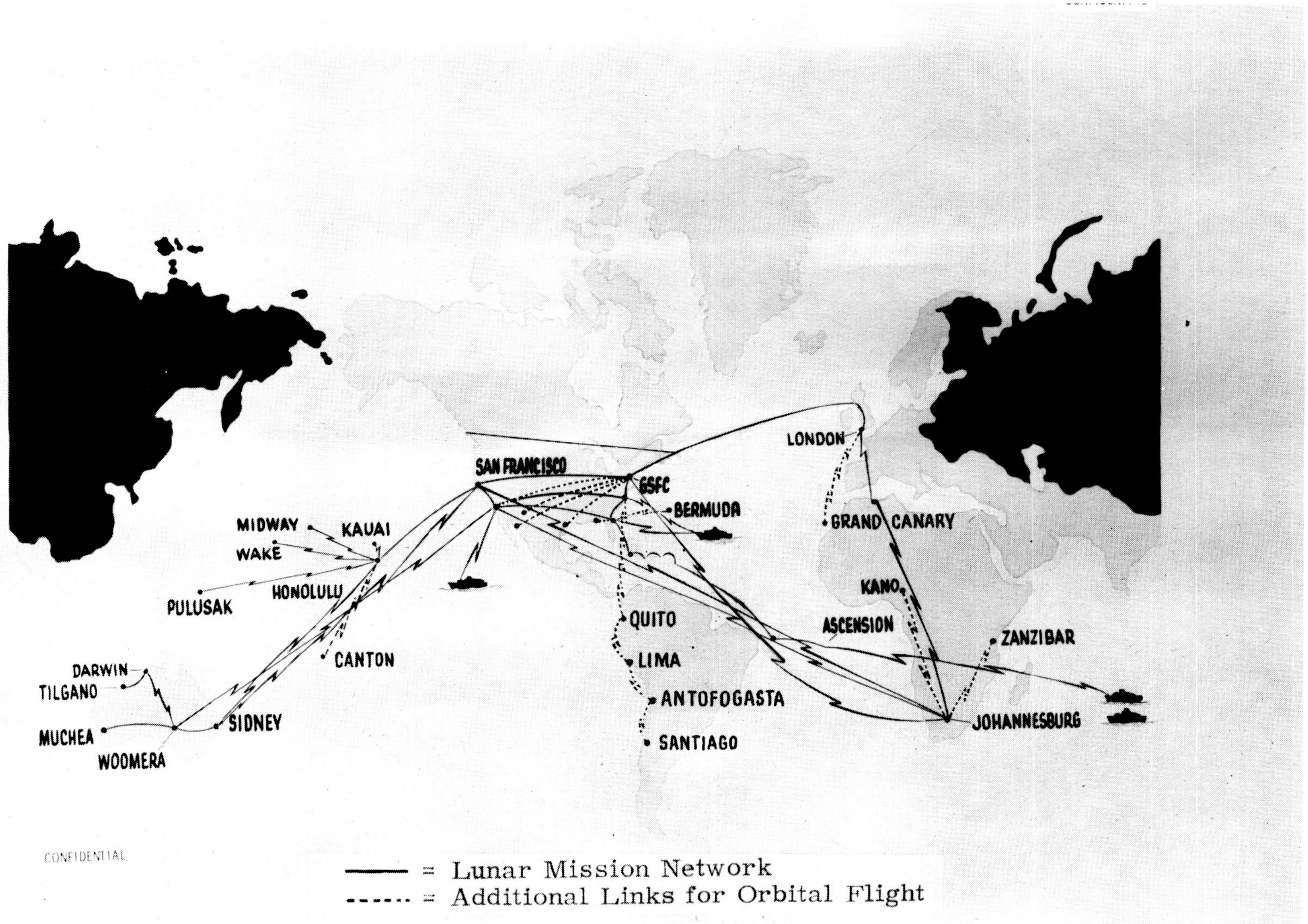
TM 25-3


VELOCITY VS GROUND RANGE
PARABOLIC PATH
REENTRY
VEHICLE ALTITUDE = 10,000 MILES
VELOCITY (x10⁴ ft/sec)
GROUND RANGE (DISTANCE FROM REENTRY) (x10³ MILES)
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
4.0
0
1.0
2.0
3.0
4.0
5.0
6.0
7.0
8.0

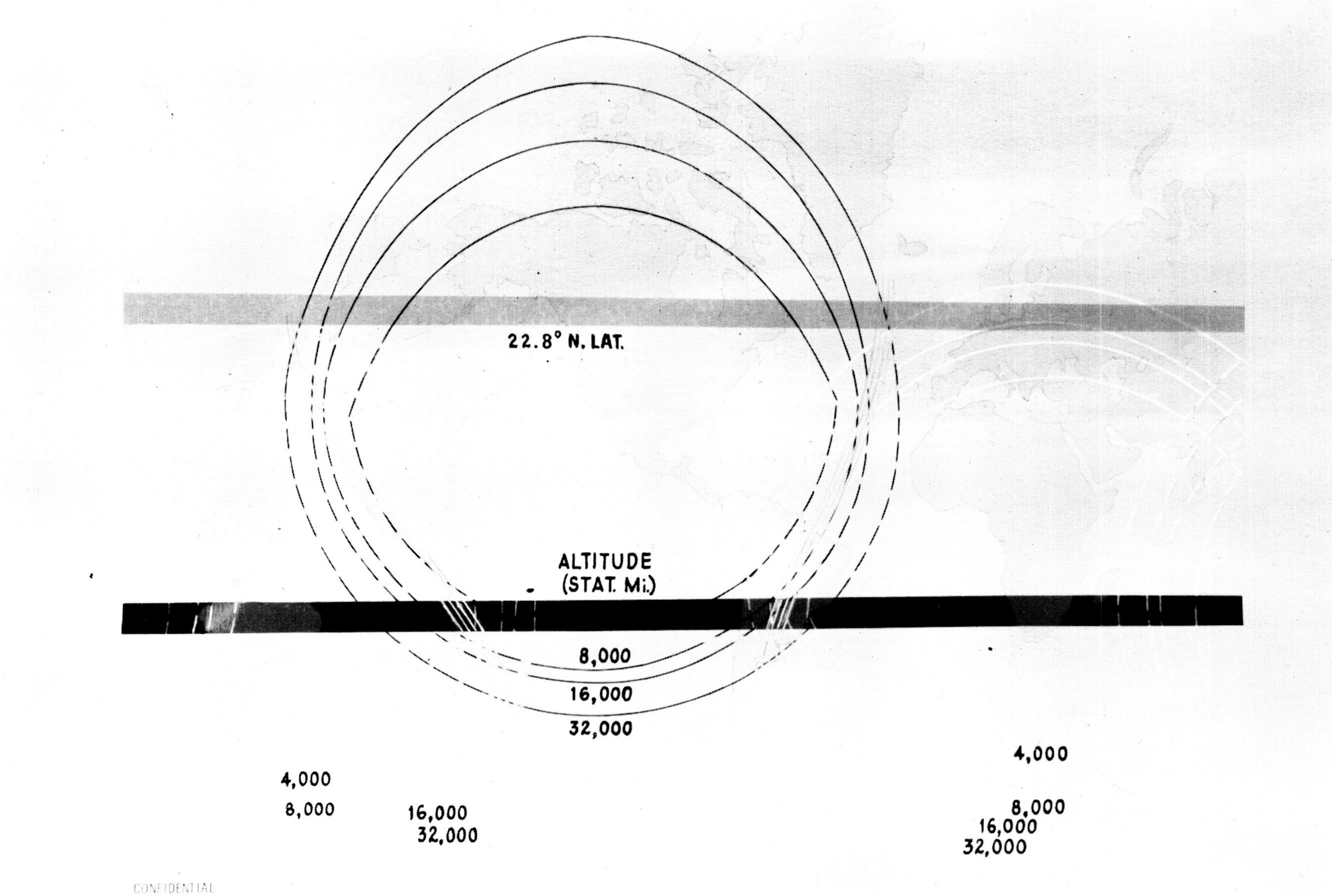
TM 25-4



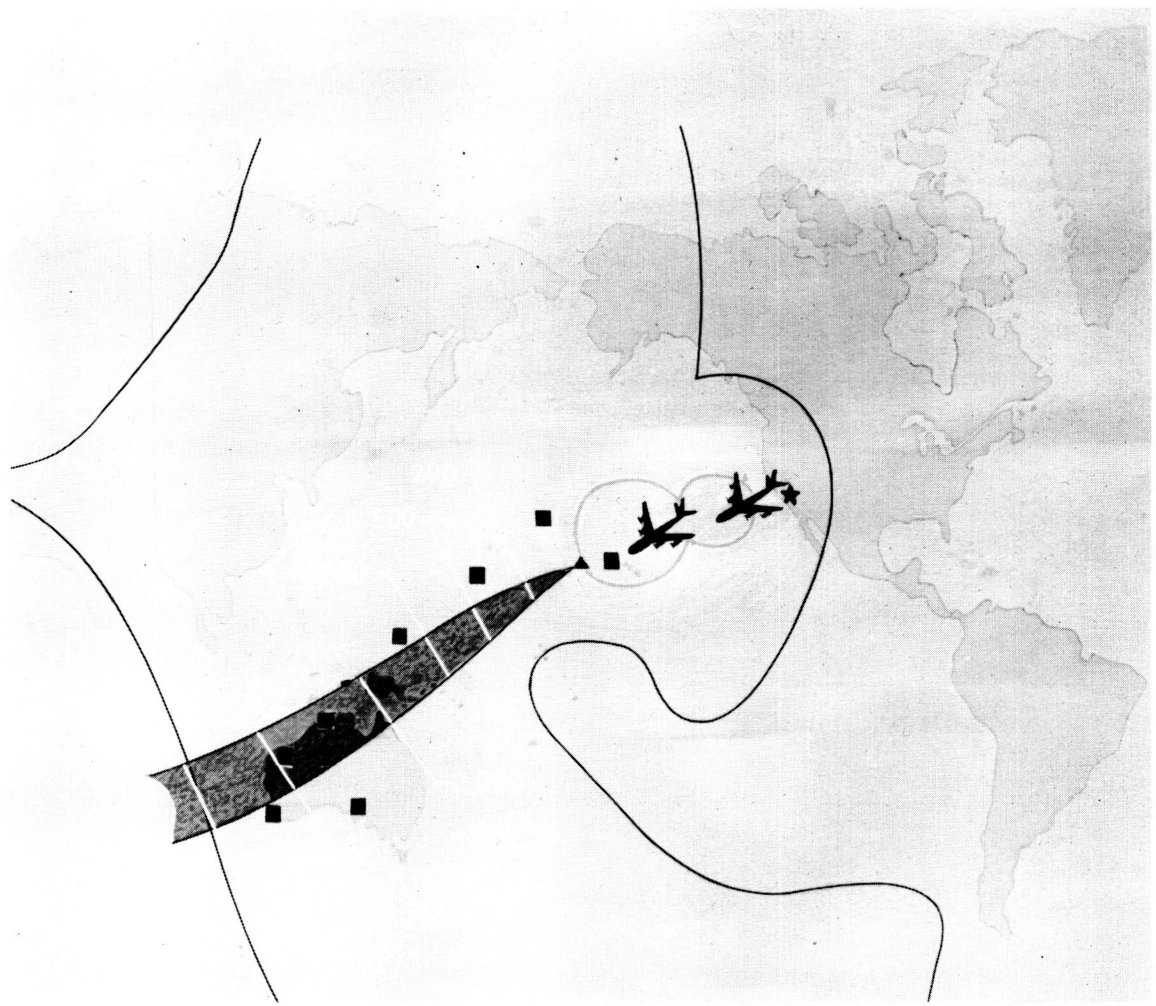
Apollo Ground Station



Ground Communication Complex

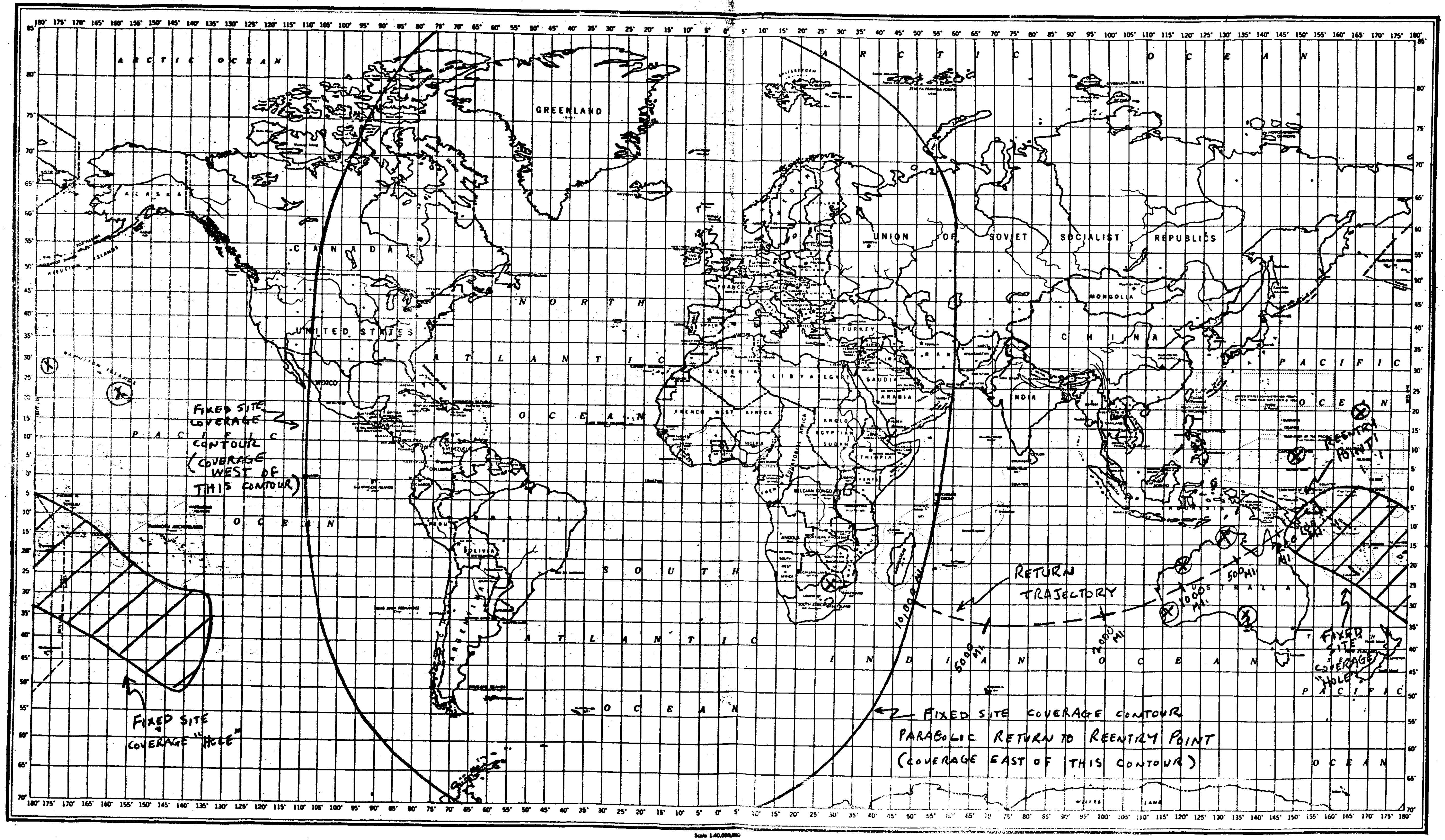


Deep Space Tracking Coverage

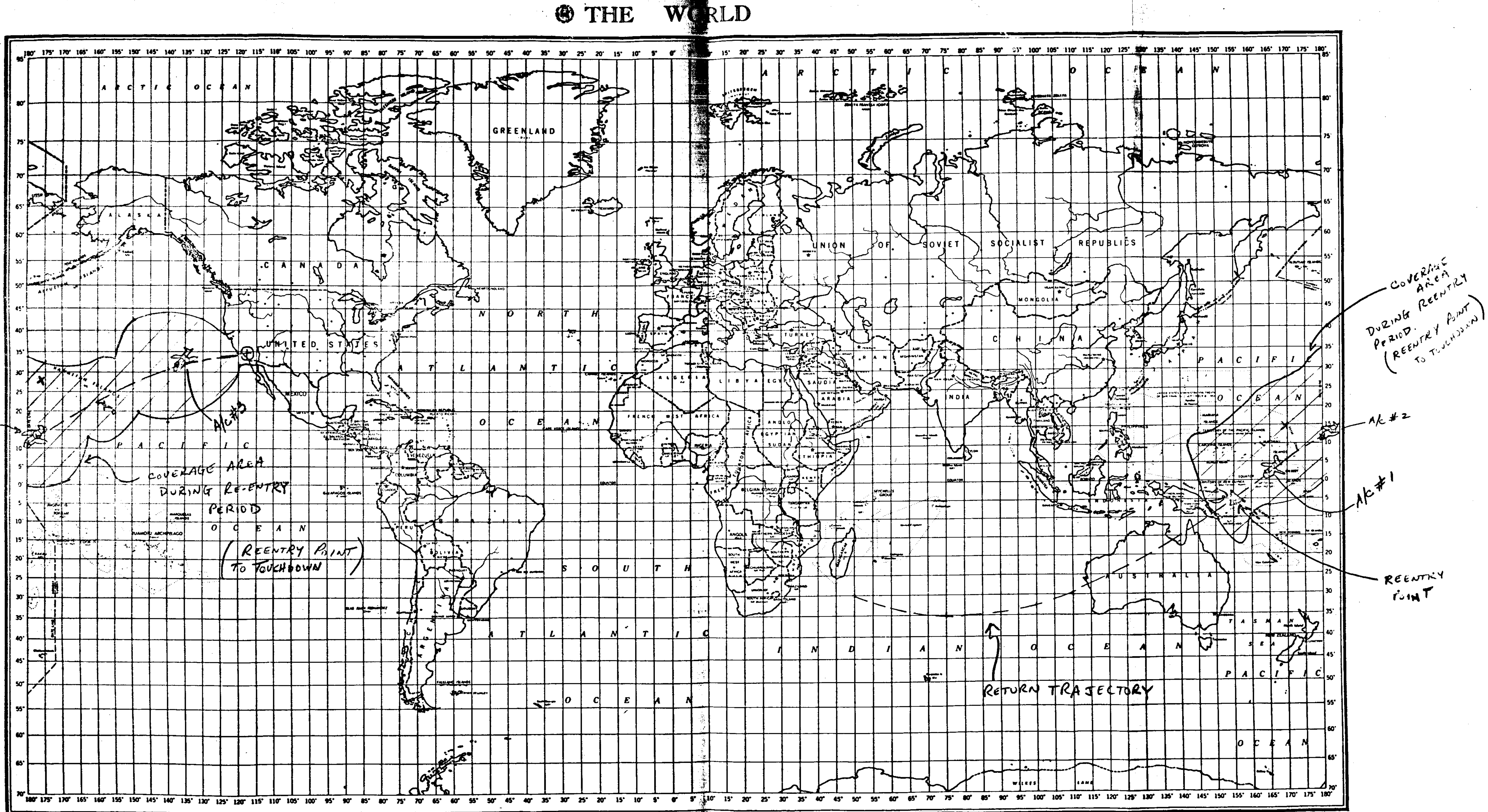


Re-entry Tracking

THE WORLD



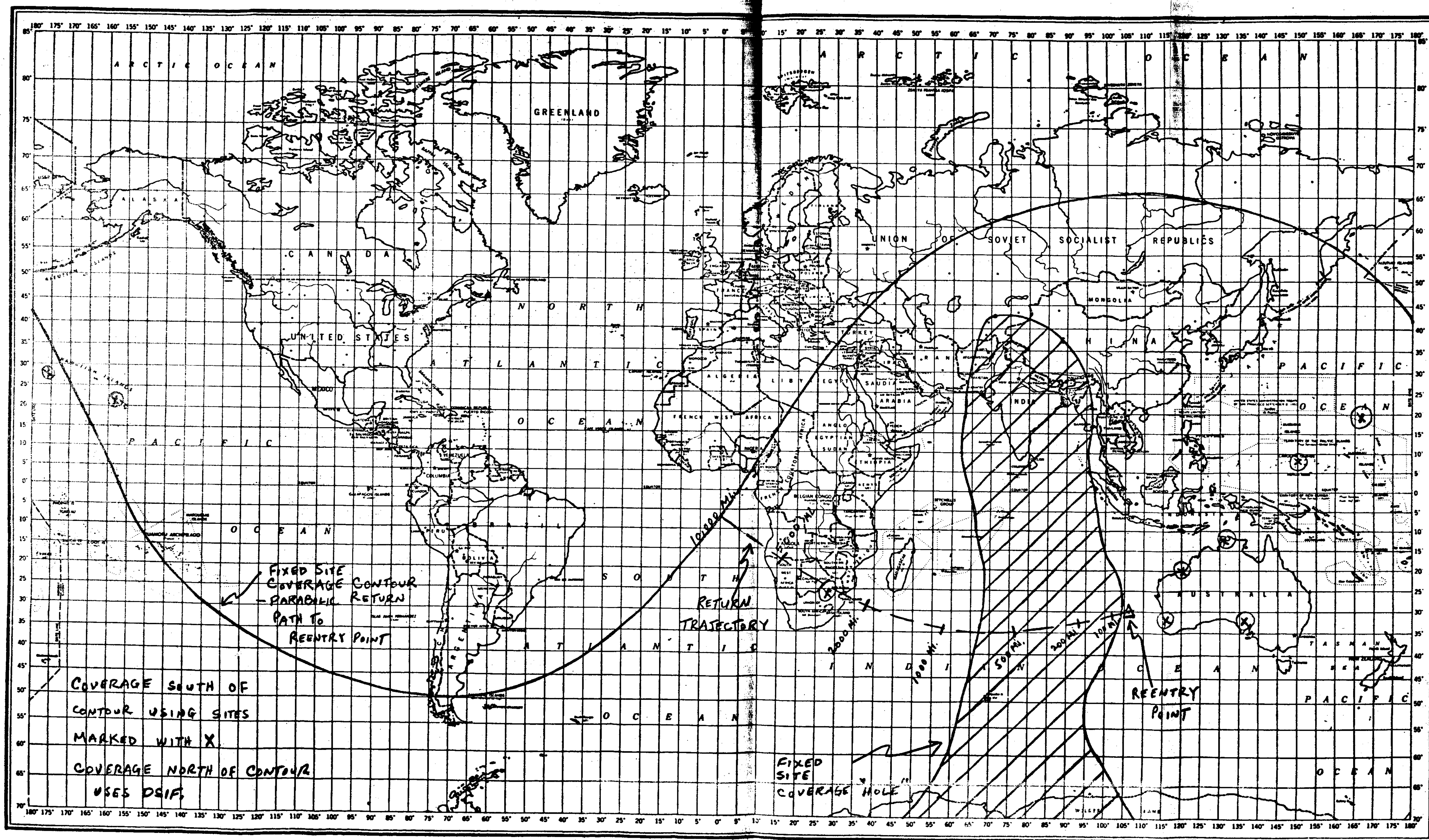
REENTRY TRACKING - ZERO DEGREE DECLINATION OF MOON



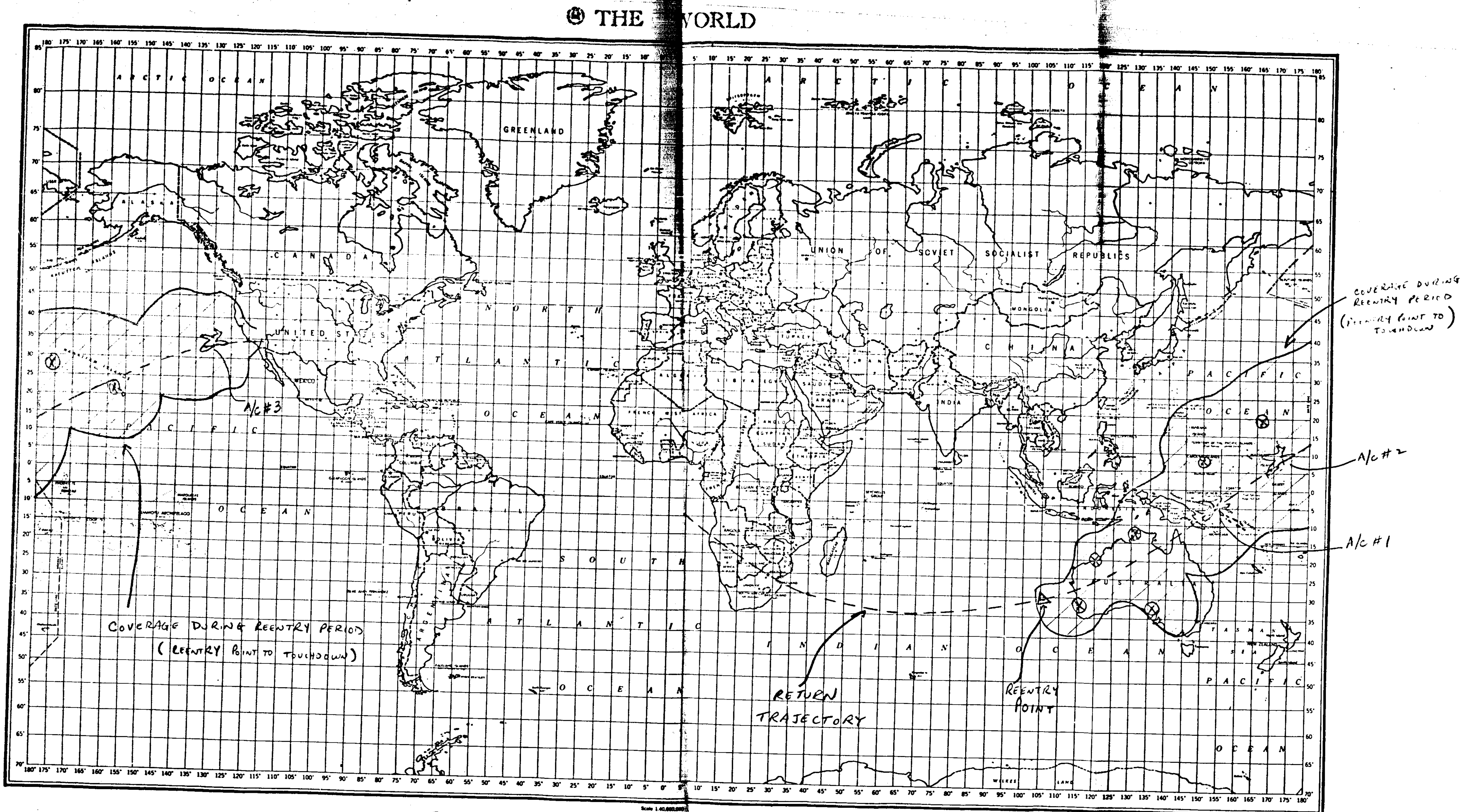
REENTRY TRACKING – ZERO DEGREE DECLINATION OF MOON
(REENTRY POINT TO TOUCHDOWN)

TM 25-10-a

THE WORLD



REENTRY TRACKING - MOON AT MAXIMUM NORTHERLY DECLINATION



REENTRY TRACKING - MOON AT MAXIMUM NORTHERLY DECLINATION
(REENTRY POINT TO TOUCHDOWN)

TM 25-11-a

Earth Orbital Mission Coverage

Pass	Station MR 1	MR 2	MR 3	MR 4	MR 5	MR 6	MR 7	MR 8	MR 9	MR 11	MR 12	MR 13	MR 14	MR 15	MR 16	MR 17	MI 5	MI 6	MI 7	MI 8	MI 11	Total Contacts Per Pass
1	x	x	x	x	x	x	x	x	x	x			x	x	x	x						14
2	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x						16
3	x	x	x				x	x			x	x	x	x	x	x					x	12
4	x	x					x				x	x	x	x	x	x					x	10
5	x						x				x	x	x	x	x						x	8
6							x				x						x	x	x		x	6
7											x							x	x	x	x	5
8						x	x	x			x										x	5
9						x													x	x		3
10							x	x		x												3
11					x		x	x		x												4
12					x													x	x			3
13				x													x	x				3
14			x	x					x								x					4
15			x	x				x	x													4
16	x	x	x	x				x	x				x		x	x						9
Contacts Per Station Per Series	6	5	6	6	4	4	9	8	5	4	7	4	6	5	6	5	3	4	4	2	6	109 / 109

Tracking Stations--Description

	Missions					Station Capabilities			
			Cislunar, Circumlunar and Lunar Orbit			Tracking			
Station Location	Sub-orbital	Orbital	Launch and Insection Phase	Deep Space Penetration Phase	Re-entry Phase	C-band	S-band	Other	Telemetry and Vehicle Communications
Cape Canaveral, Florida	x	x	x			x	x		x
Grand Bahama Island	x	x	x			x	x		x
Grand Turk Island	x	x				■	x		x
San Salvador Island	x		x			x	x		x
Puerto Rico Island	x		x				x		x
Antigua Island	x		x			x	x		x
Fernando de Noronha Island	x		x				x		x
Ascension Island	x		x			x	x		x
Bermuda		x				x	x		x
Grand Canary Island		x					x		x
Kano, Nigeria		x							x
Zanzibar		x							x
Muchea, Australia	x	x	x		x		x		x

Tracking Stations--Description (continued)

Station Location	Missions					Station Capabilities			
			Cislunar, Circumlunar and Lunar Orbit			Tracking			
	Sub-orbital	Orbital	Launch and Insection Phase	Deep Space Penetration Phase	Re-entry Phase	C-band	S-band	Other	Telemetry and Vehicle Commu-nications
Woomera, Australia	x	x	x	x	x	x	x		x
Canton Island		x							x
Pt. Arguello, California		x				x	x		x
Kauai Island, Hawaii	x	x	x		x	x	x		x
Guayamas, Mexico		x					x		x
White Sands, New Mexico		x				x			
Corpus Christi, Texas		x					x		x
Eglin AFB, Florida		x				x	[redacted]		
Antofogasta, Chile		x						x	x
Lima, Peru		x						x	x
Santiago, Chile		x						x	x
Johannesburg, South Africa	x	x	x	x		x	x	x	x
Quito, Ecuador		x						x	x

[redacted]

Tracking Stations--Description (continued)

	Missions					Station Capabilities			
		Cislunar, Circumlunar and Lunar Orbit				Tracking			
Station Location	Sub-orbital	Orbital	Launch and Insection Phase	Deep Space Penetration Phase	Re-entry Phase	C-band	S-band	Other	Telemetry and Vehicle Commu-nications
Goldstone, California				x			x		x
Tilgano, Australia	x		x		x	x			x
Darwin, Australia	x		x		x	x			x
Pulusak Island	x		x		x	x			x
Wake Island	x		x		x	x			x
Midway Island	x		x		x	x			x
Edwards AFB, California	x	x			x		x		x
Shipboard Stations		x	x			x			x
Airborne Stations	x				x			x	x

[redacted]

APOLLO COMPUTATION FACILITIES

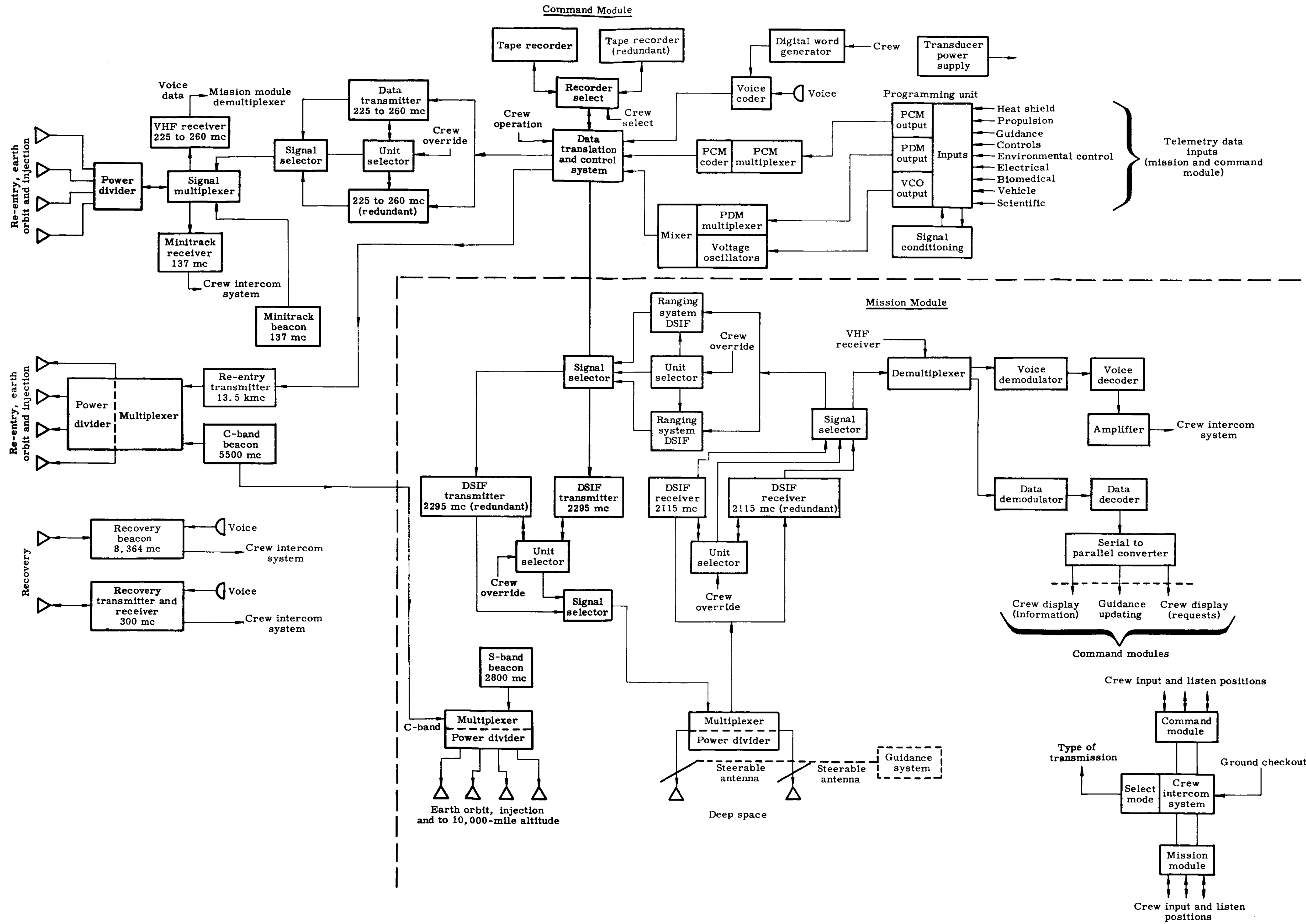
TRACKING NET	COMPUTER LOCATION	COMPUTER TYPE	NUMBER OF COMPUTERS
ATLANTIC MISSILE RANGE	CAPE CANAVERAL, FLORIDA	IBM 709	2
MERCURY NET	CAPE CANAVERAL BERMUDA GODDARD SFC	SEE ABOVE IBM 709 IBM 7090	SEE ABOVE 1 2
DEEP SPACE INSTRUMENTATION FACILITY	PASADENA[1] LOS ANGELES[2] (STL)	IBM 704 IBM 7090	1 2
PACIFIC MISSILE RANGE	PT. ARGUELLO, CALIF.[3] HAWAII[4]	IBM 709 IBM 709	1 1

1 – PLANNING SHIFT TO IBM 7090 IN THE FUTURE

2 – LEASED FACILITY USED FOR BACKUP

3 – CURRENTLY BEING INSTALLED – READY DEC. 1961

4 – CONTEMPLATED FOR FUTURE – PRESENTLY UNFUNDED

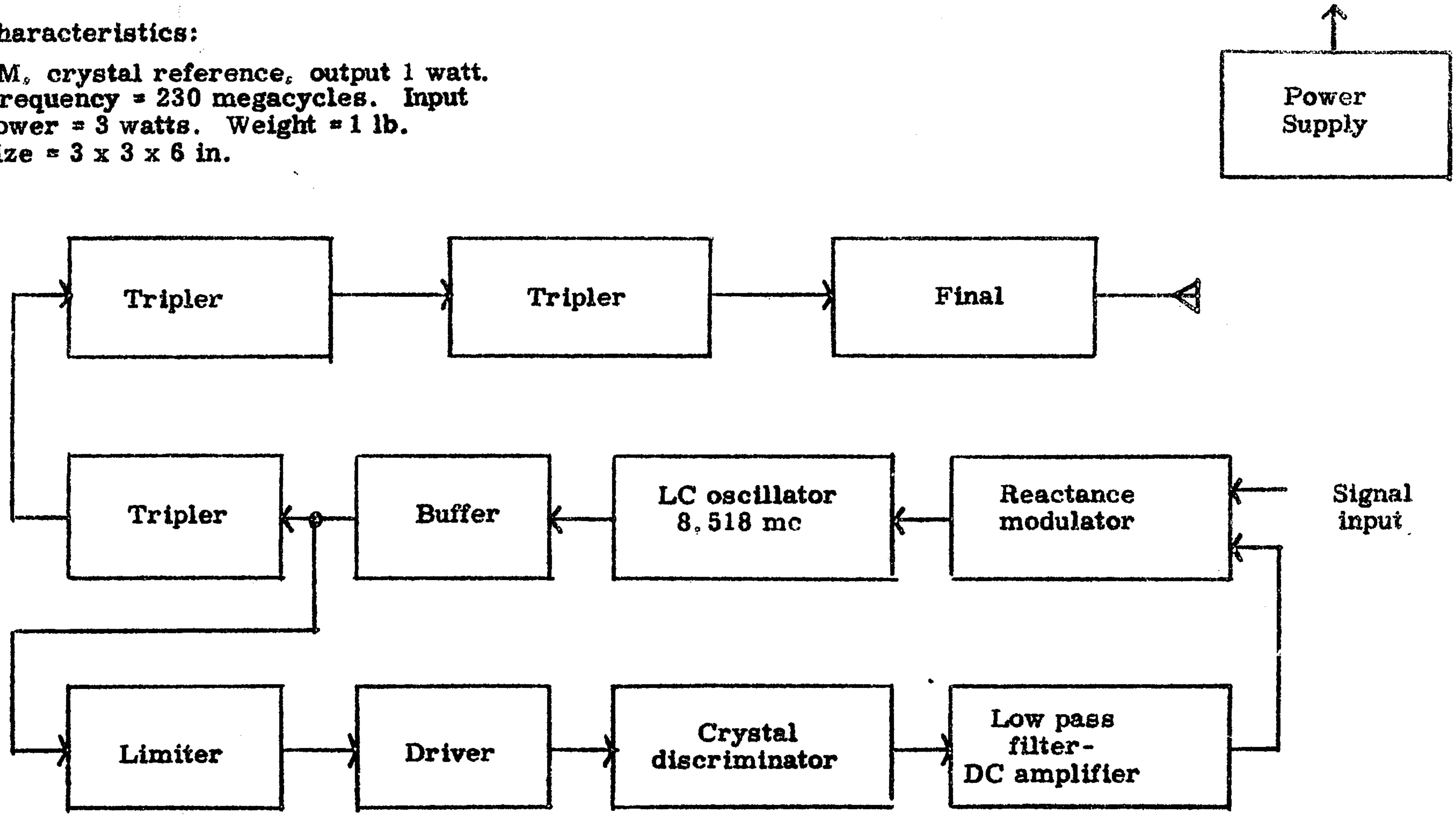


Onboard Tracking, Communication and Instrumentation System

TM 25-16

Characteristics:

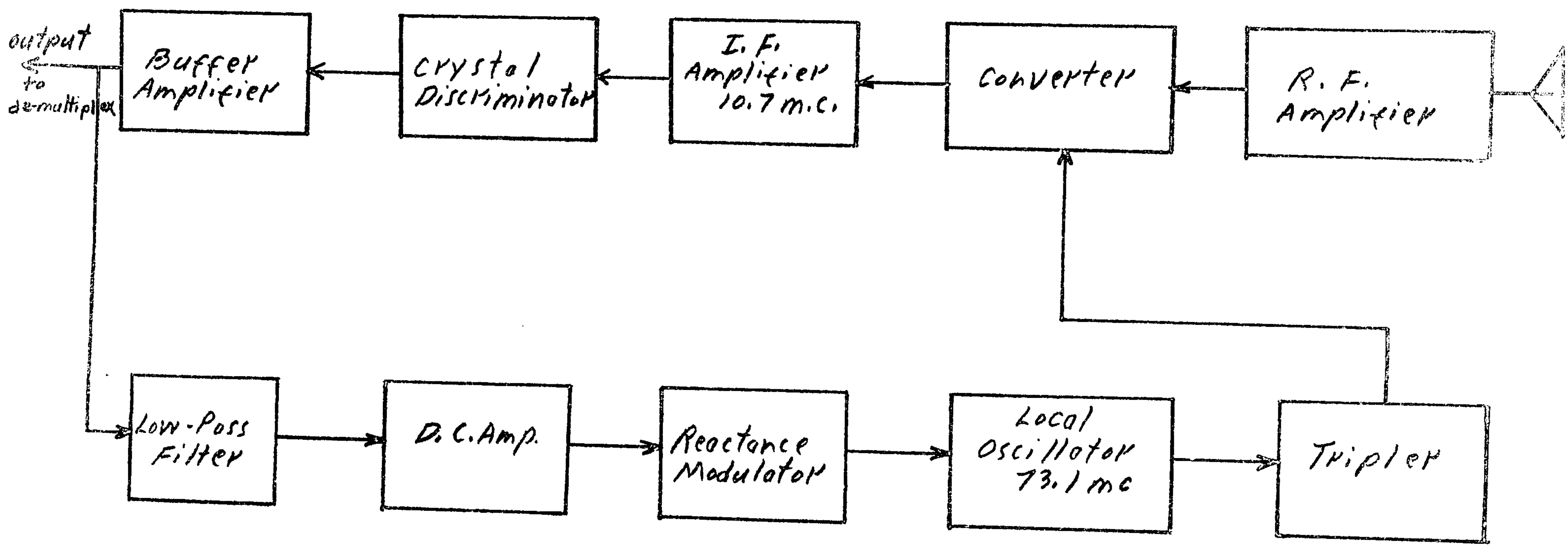
FM, crystal reference, output 1 watt. Frequency = 230 megacycles. Input power = 3 watts. Weight = 1 lb. Size = 3 x 3 x 6 in.



Apollo--VHF Transmitter.

VHF Receiver

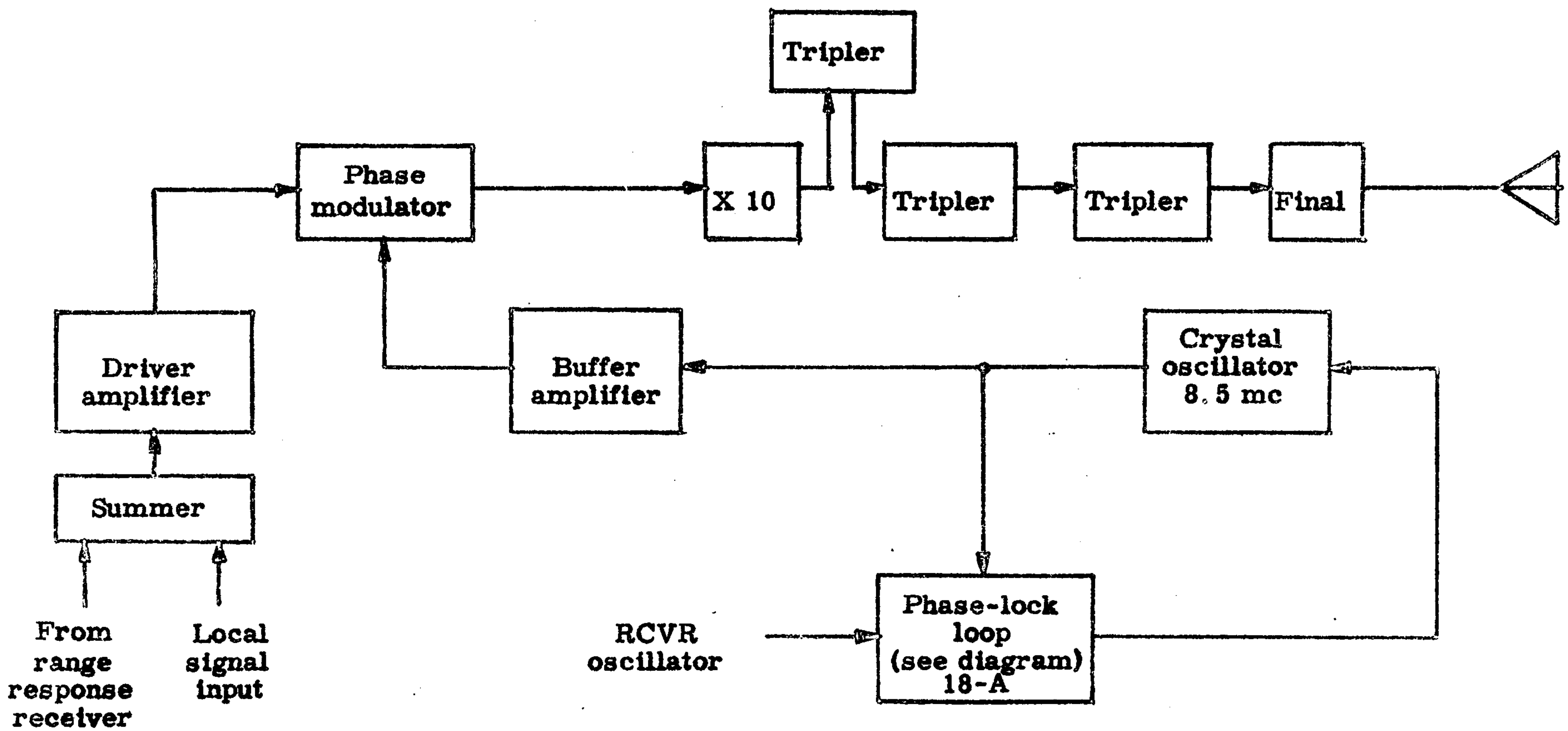
Characteristics: F.M. - Crystal Controlled - 230 megacycles
Size 1"X 3"X 6" - Wt. 0.5 lbs. - Power in 0.25 w. - Doppler Track - AFC



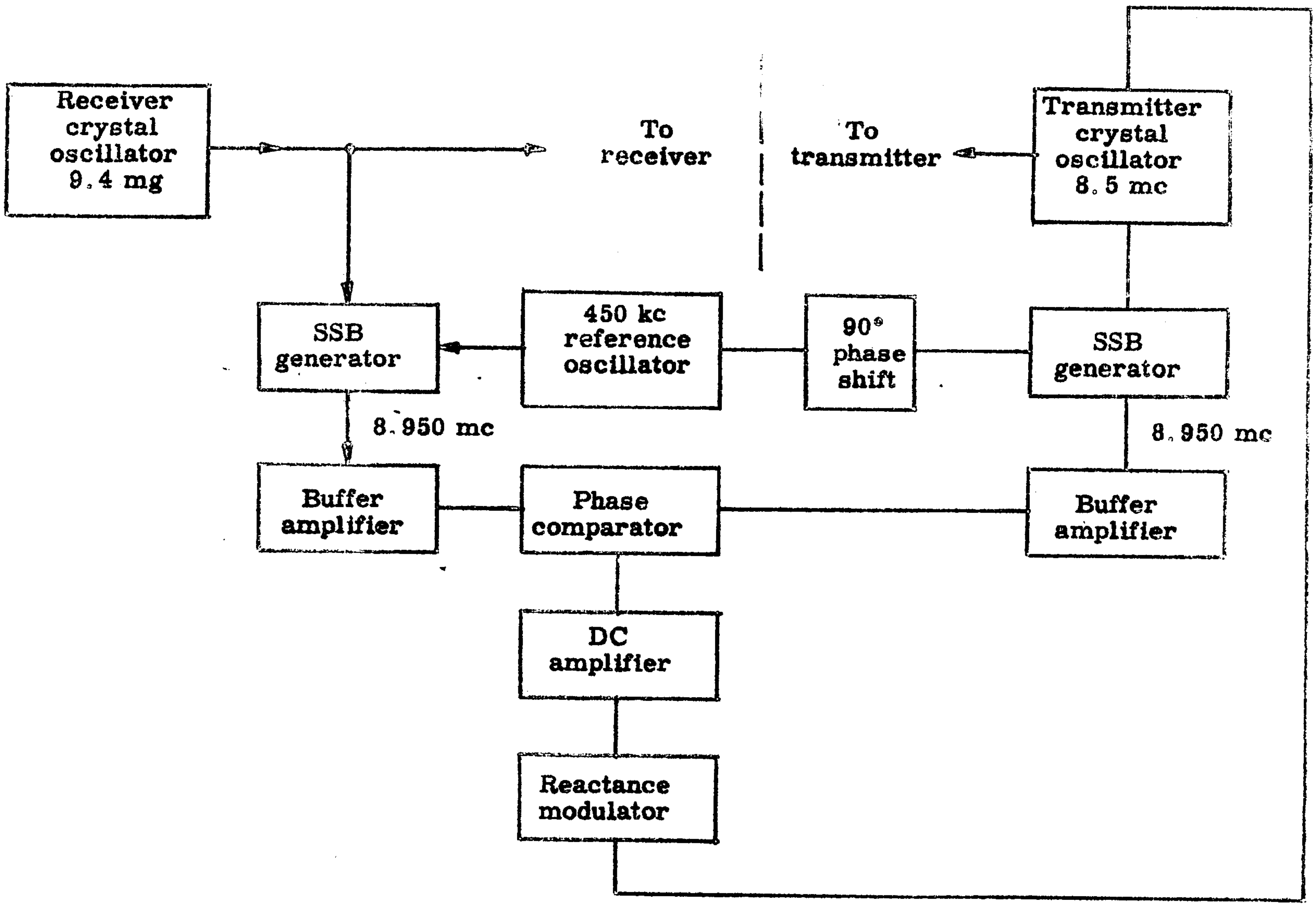
TM 25-18

Characteristics:

PM, crystal control, doppler locked, 10-watt output
Frequency = 2295 megacycles. Dual unit size =
3 x 6-1/2 x 13 in. Weight = 3 lb. Input power = 30 watt



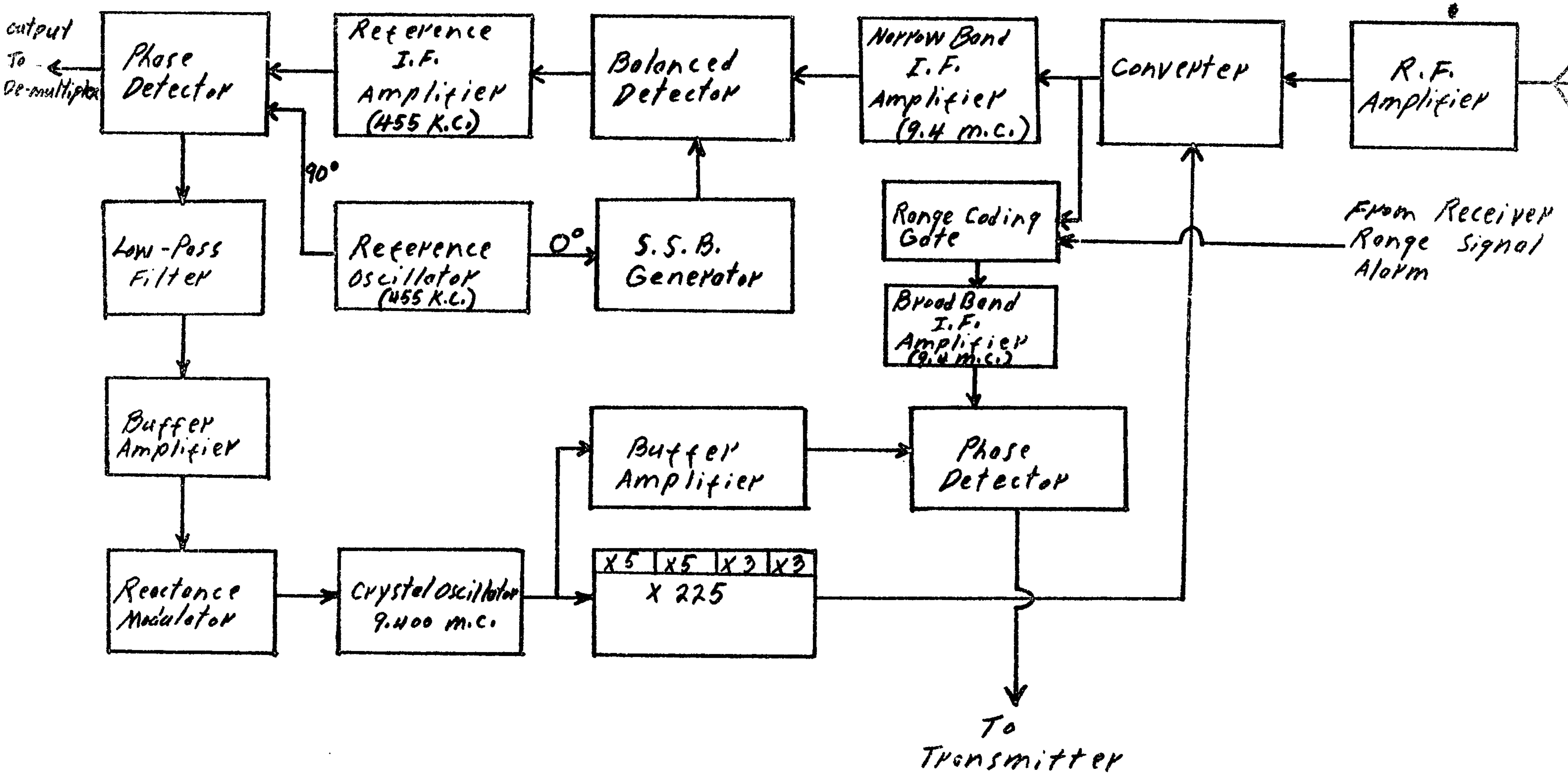
Apollo S-Band Transmitter



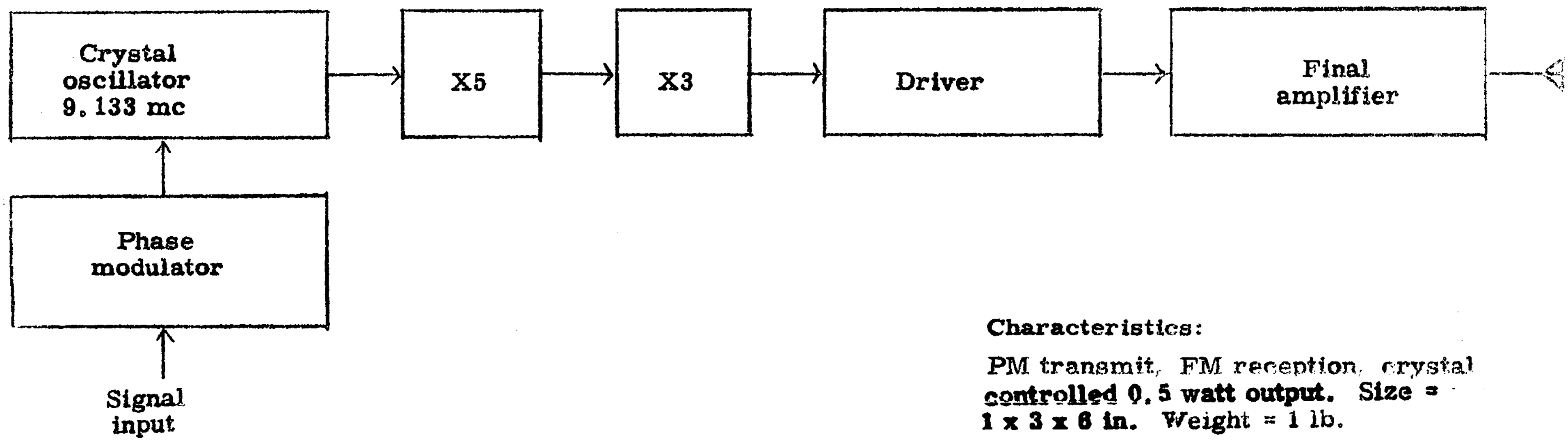
Apollo S-Band Transmitter-Receiver
Phase Lock Loop

S-Band Receiver

Characteristics: Frequency 2115 m.c. - P.M. - crystal Controlled - Double conversion Range Code Pick-off - N.B. 200 K.C. B.W. - Ref I.F. 200 K.C. B.W. - B.B. I.F 3.0 m.c. Bw Wt. 0.5 lbs. - Size 1" X 3" X 6" -

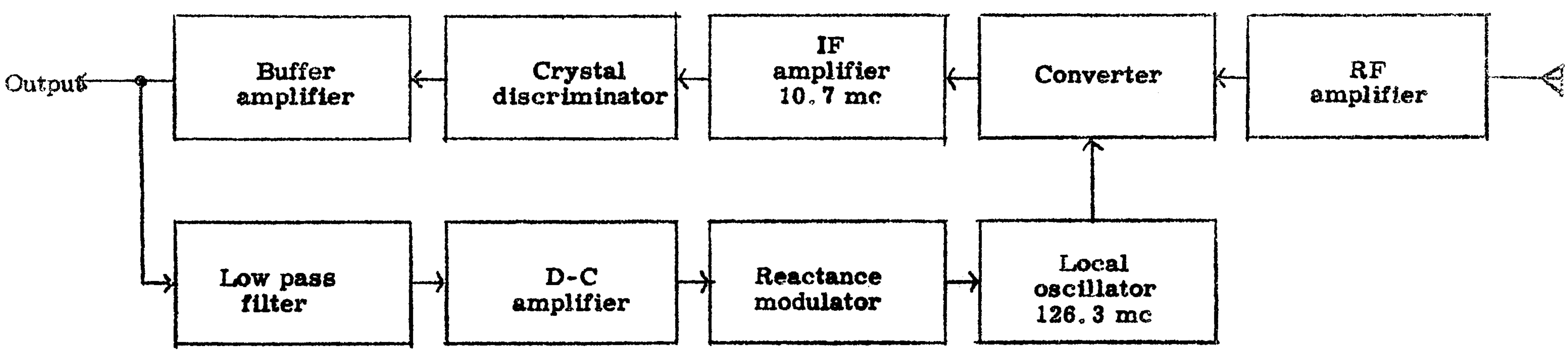


TM-25-20



Characteristics:

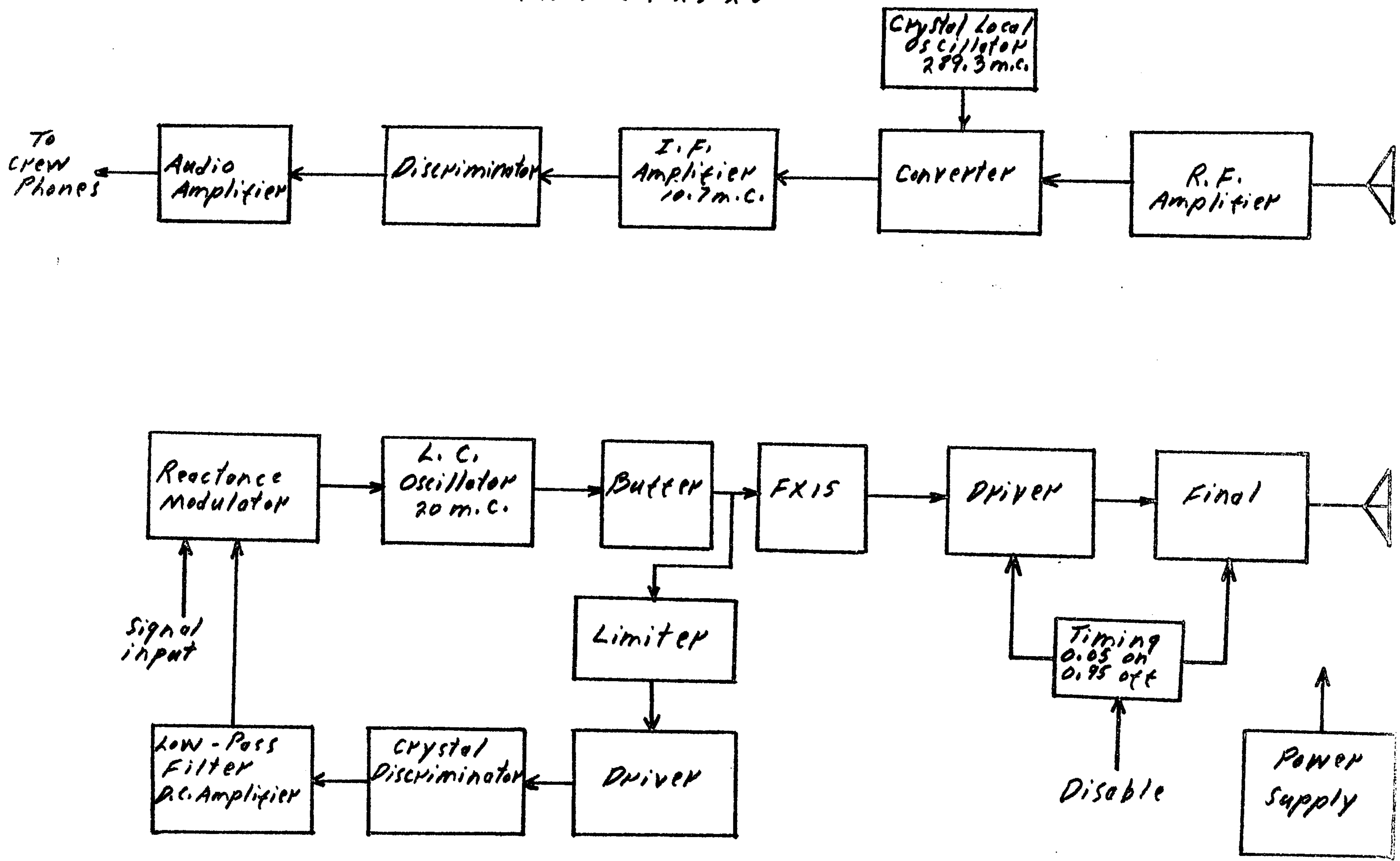
PM transmit, FM reception, crystal **controlled 0.5 watt output. Size = 1 x 3 x 6 in.** Weight = 1 lb.



Apollo--VHF Transmitter-Receiver, 137 Megacycles

UHF Transmitter-Receiver

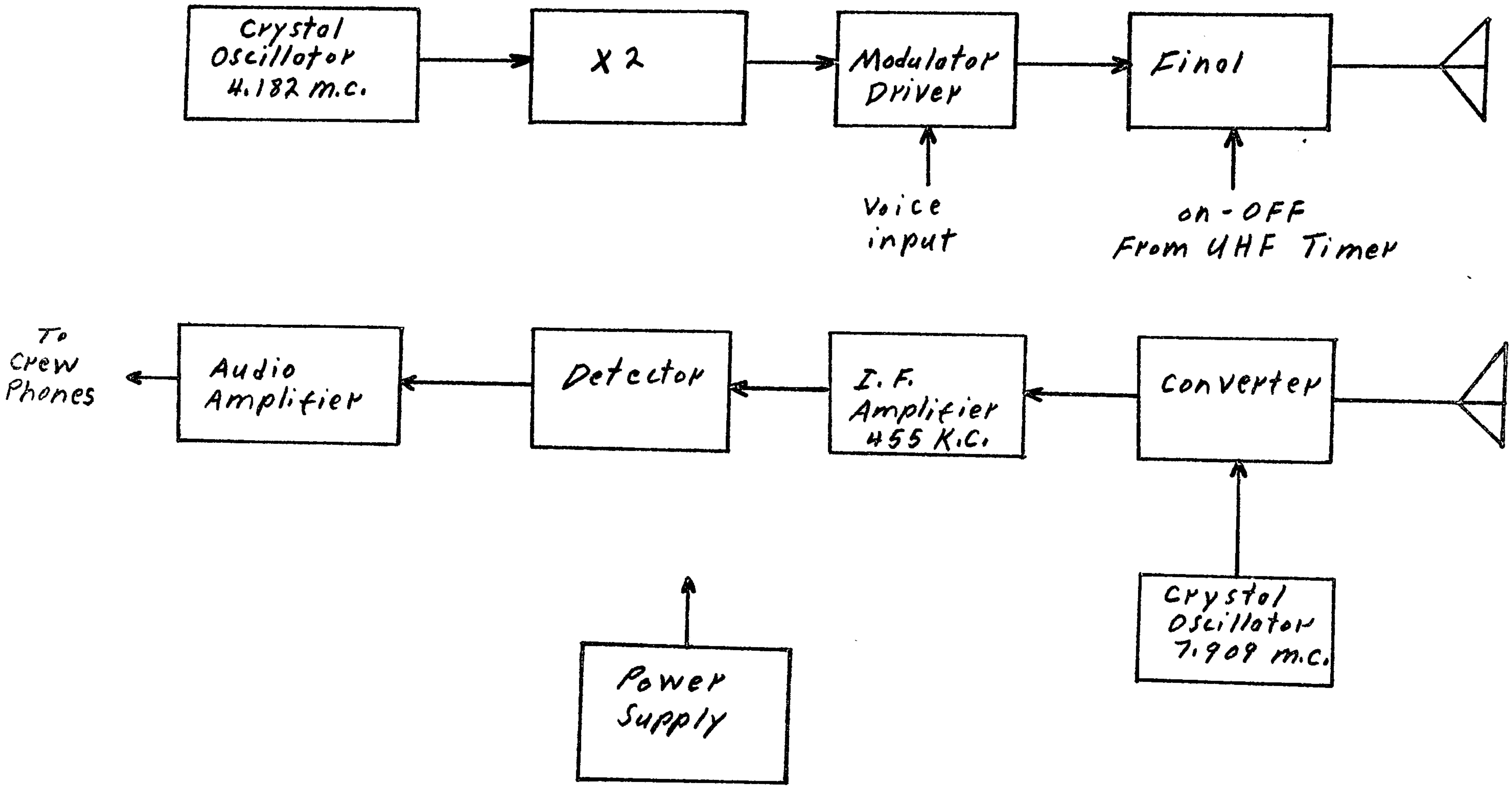
Characteristics: Modulation F.M. - Crystal Controlled - 300 m.c. - Power out 5w
I.F. Band Width 150 K.C. - Tsx. wt. 3 lbs. - Rcvr wt. .5 lbs. - Power in Tsx. 15w. Rcvr .25w
Tsx. Size 3"X6"X6" - Rcvr size 1"X3"X6" -



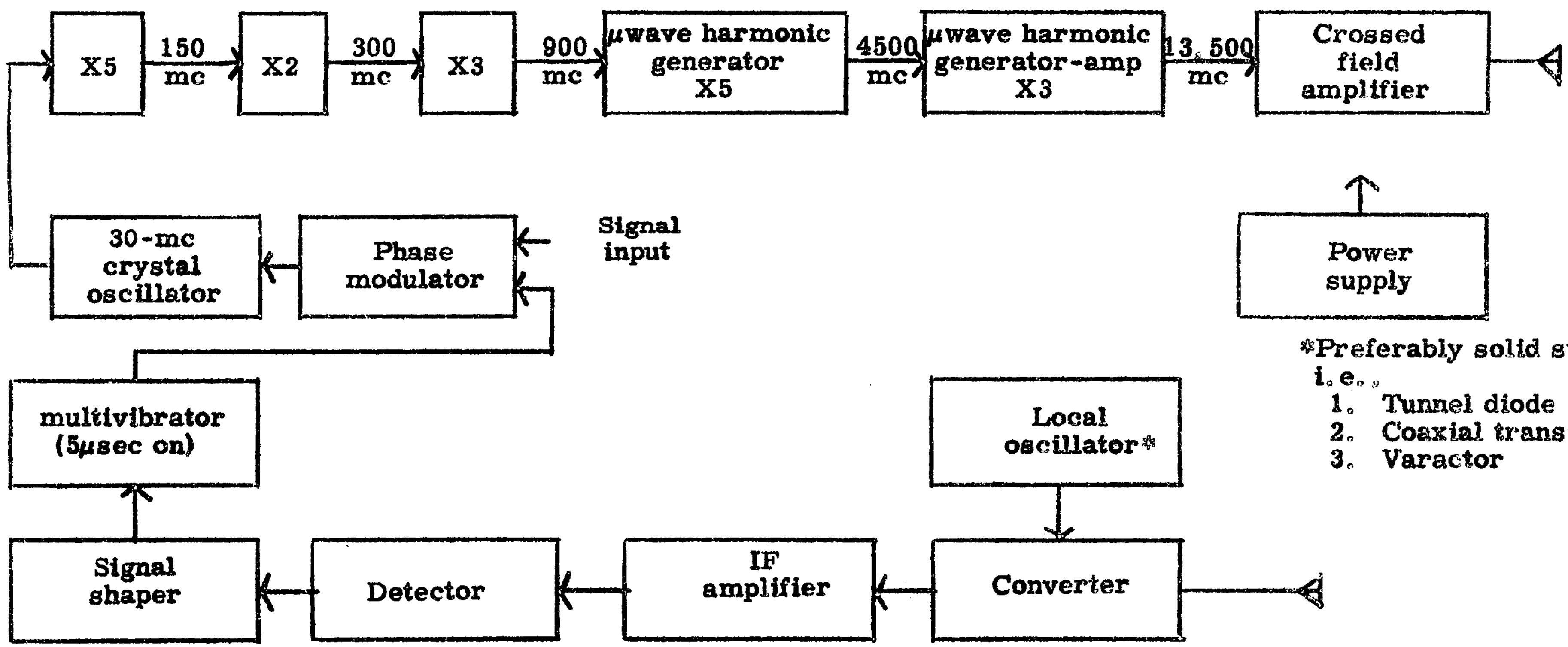
HF Transmitter - Receiver

8.364 M.C

Characteristics :- A.M. - Crystal Controlled - Power Output 5 watts
Tsx. Input Power 10 watts - Tsx. size 1" x 3" x 6" - Tsx. wt. 2 lbs. -



TMX5-23



*Preferably solid state, i.e.,
1. Tunnel diode
2. Coaxial transistor
3. Varactor

Characteristics:

FM/FM crystal reference, 50-watt output. Size=6 x 12 x 12 in. Wt = 30 lb.

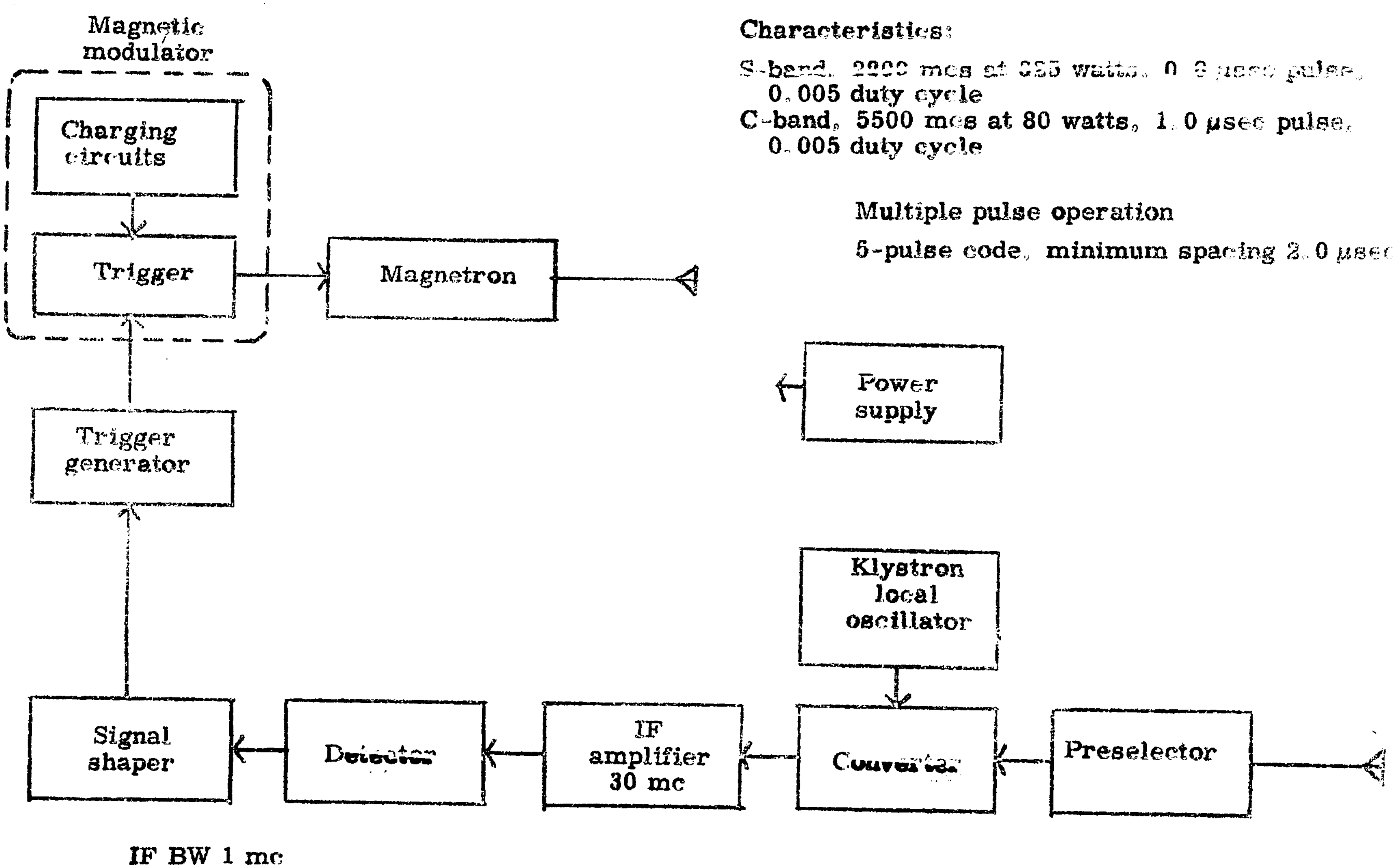
K_E-Band Transmitter--Beacon, 13,500 Megacycles

Characteristics:

S-band, 2[illegible] mcs at [illegible] watts, 0.[illegible] μsec pulse,
0.005 duty cycle
C-band, 5500 mcs at 80 watts, 1.0 μsec pulse,
0.005 duty cycle

Multiple pulse operation

5-pulse code, minimum spacing 2.0 μsec



Apollo--C and S-Band Beacons

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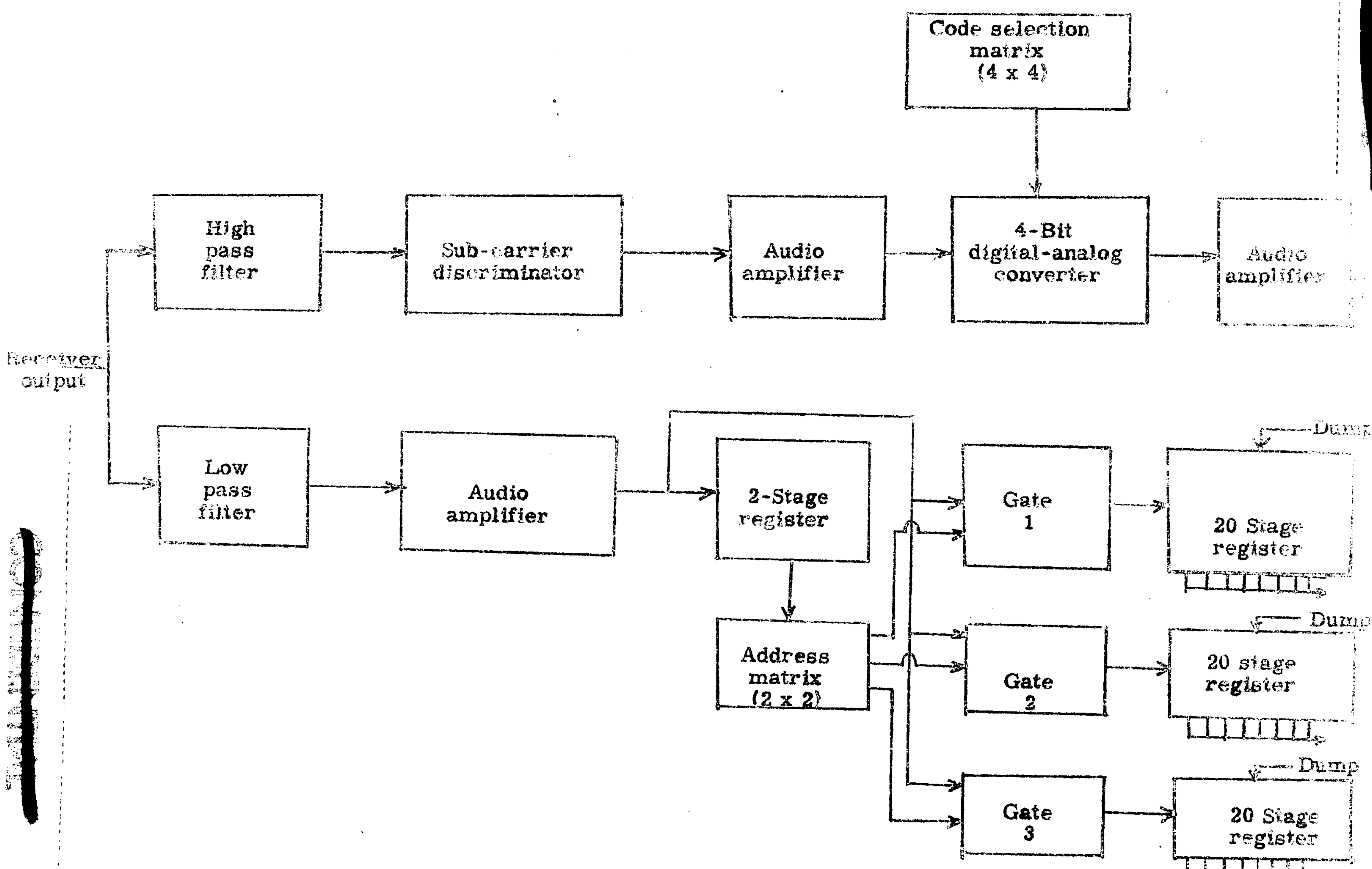
[redacted]

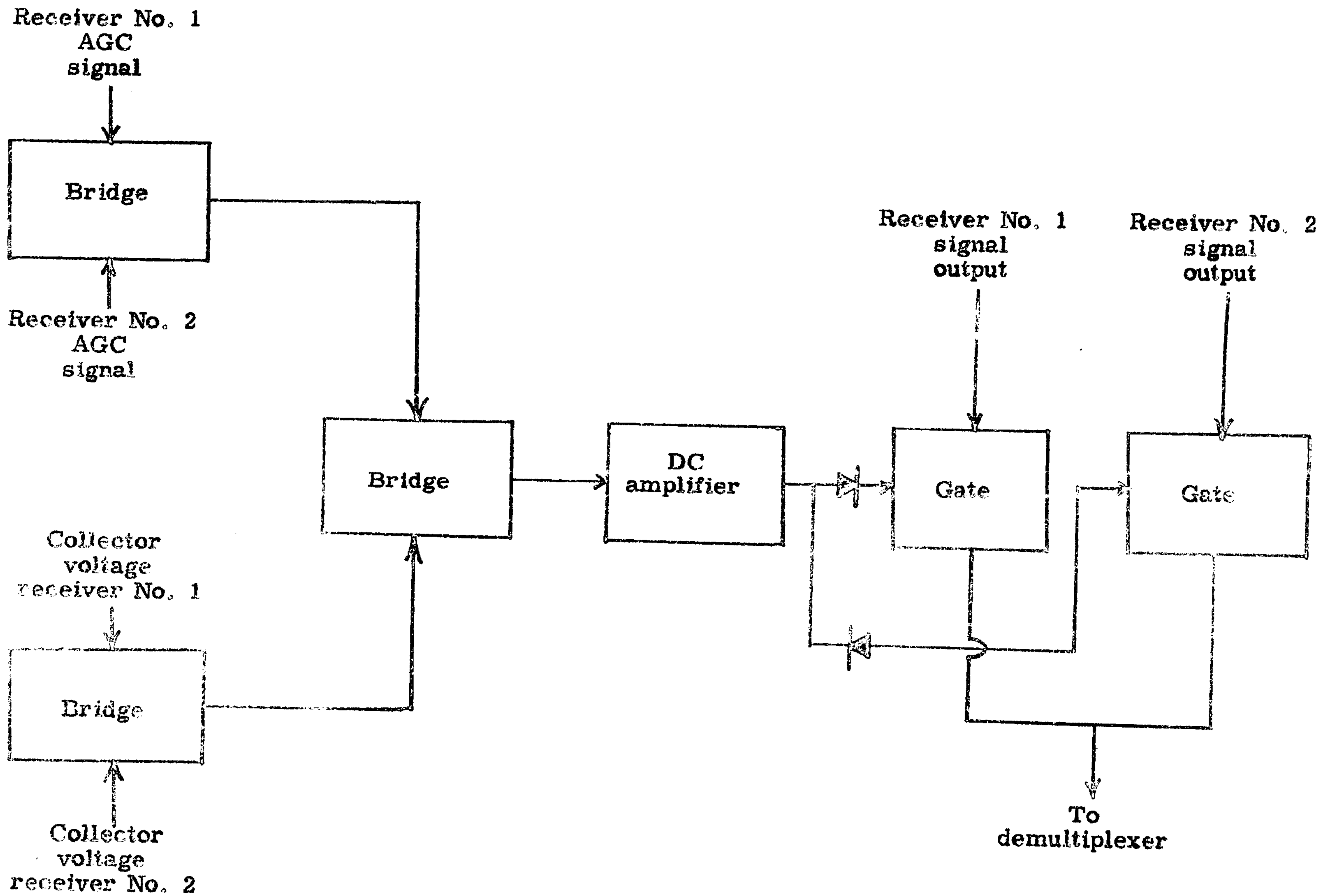


Apollo Vehicle Reception System
Demodulator-Decoder

[redacted]

TM25-26

[redacted]



Apollo-Vehicle Reception Signal Processing
Signal Selector (typical)

[redacted]

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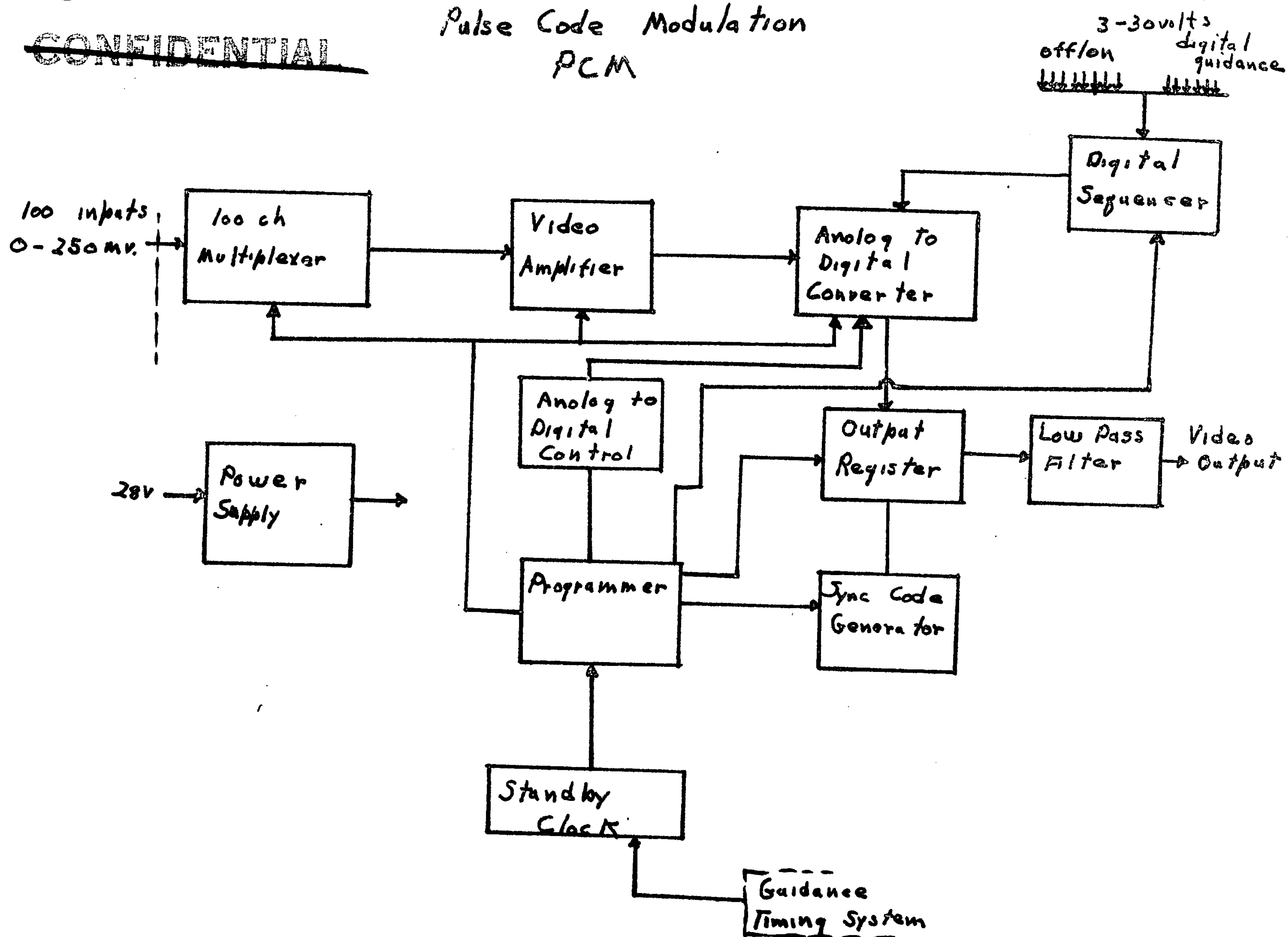
Pulse Code Modulation
PCM



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TM 25-28A

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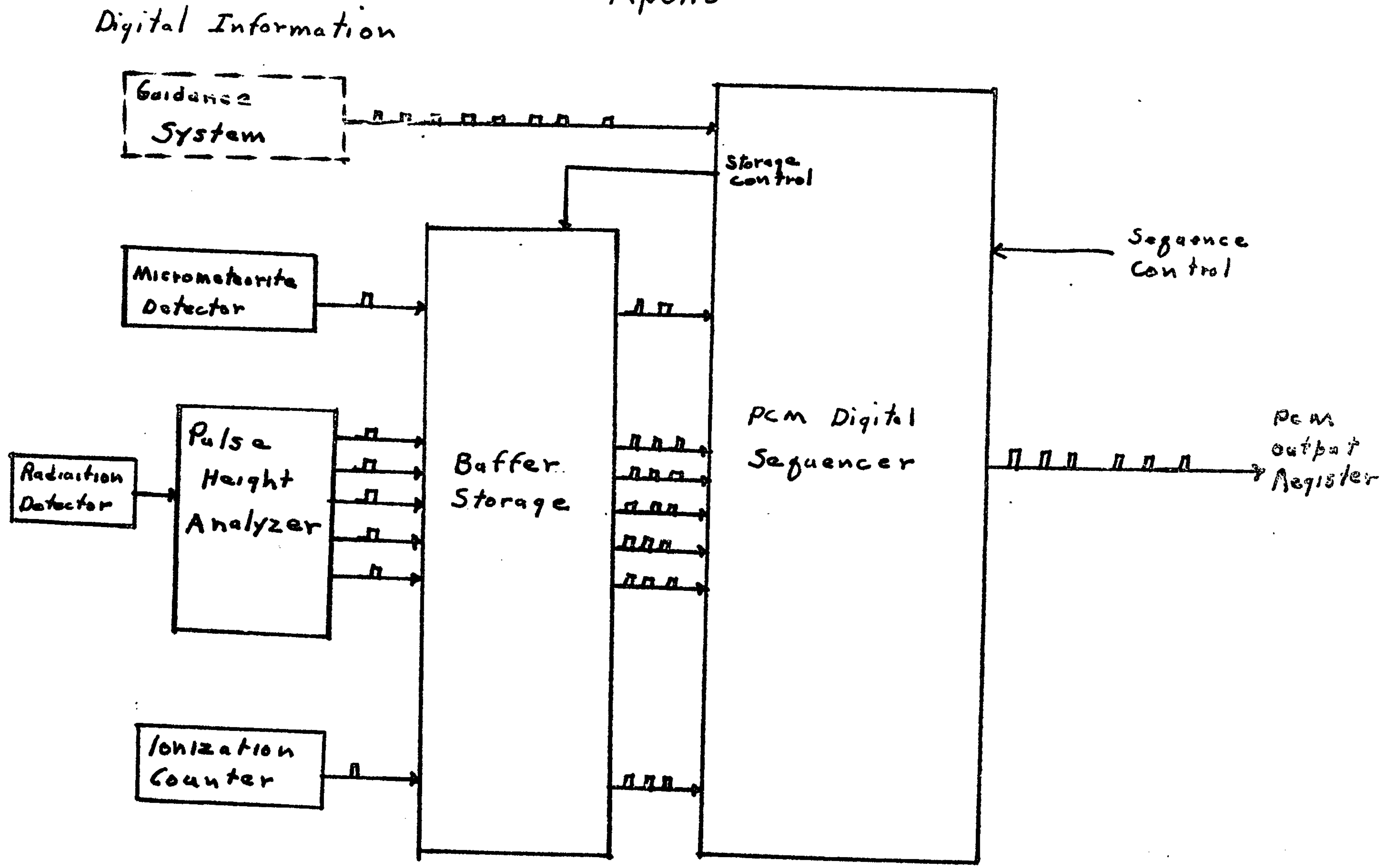
Typical Measurement System "Apollo"



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N25-28B

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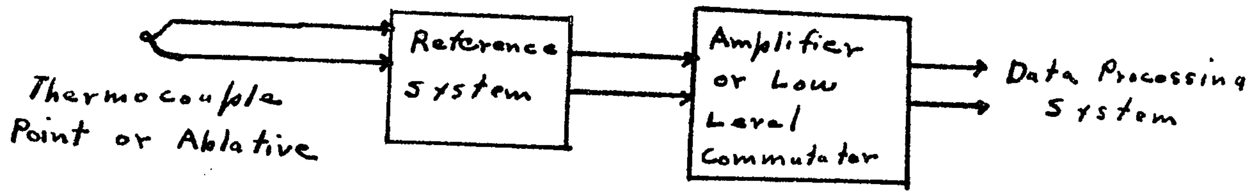
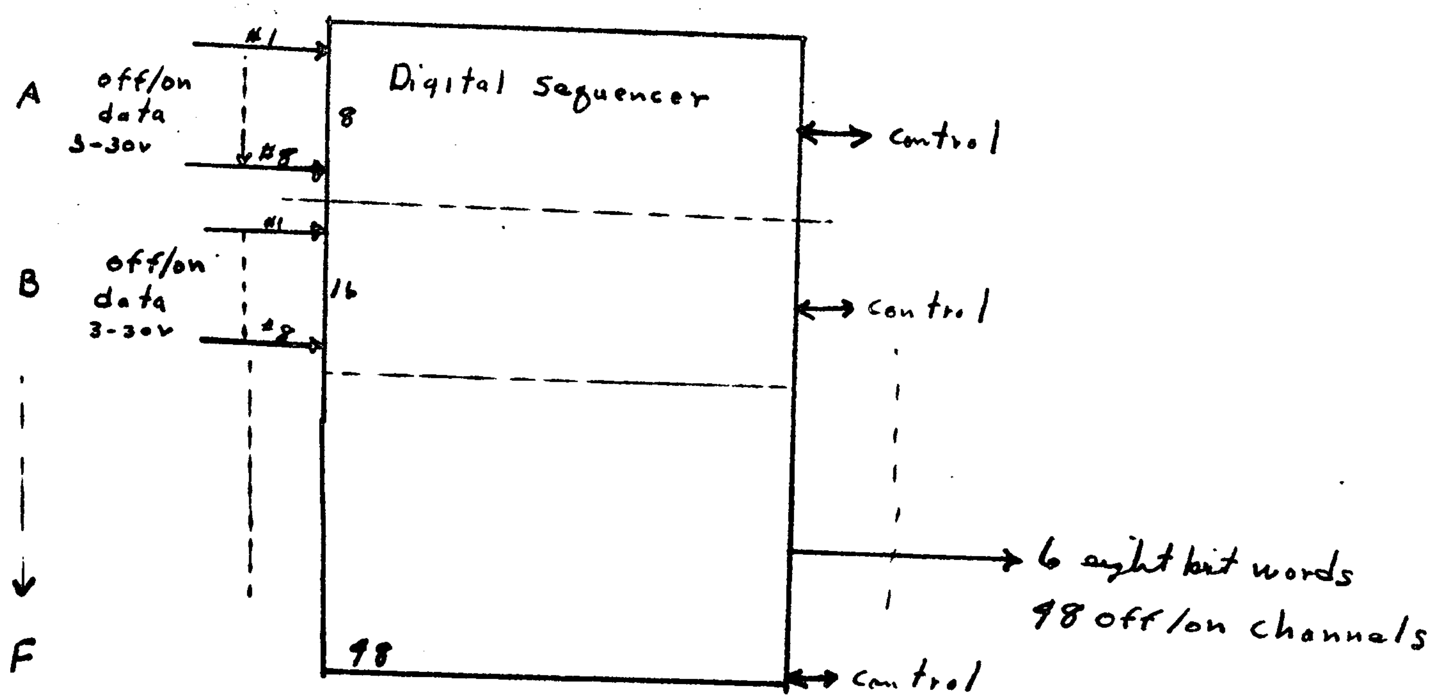
Typical Measurement Systems
"Apollo"

Temperature 1000°F – 5000°F



off/on Info



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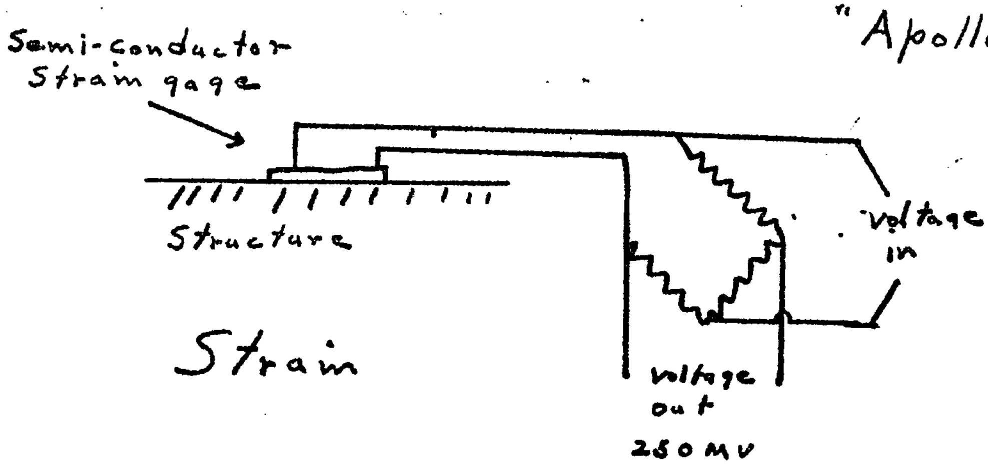
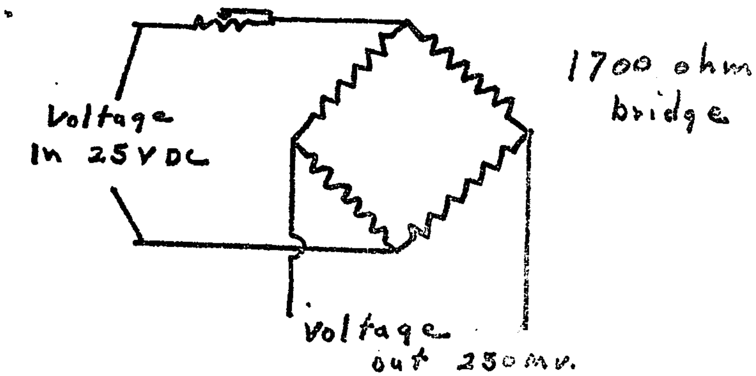
TM25-28

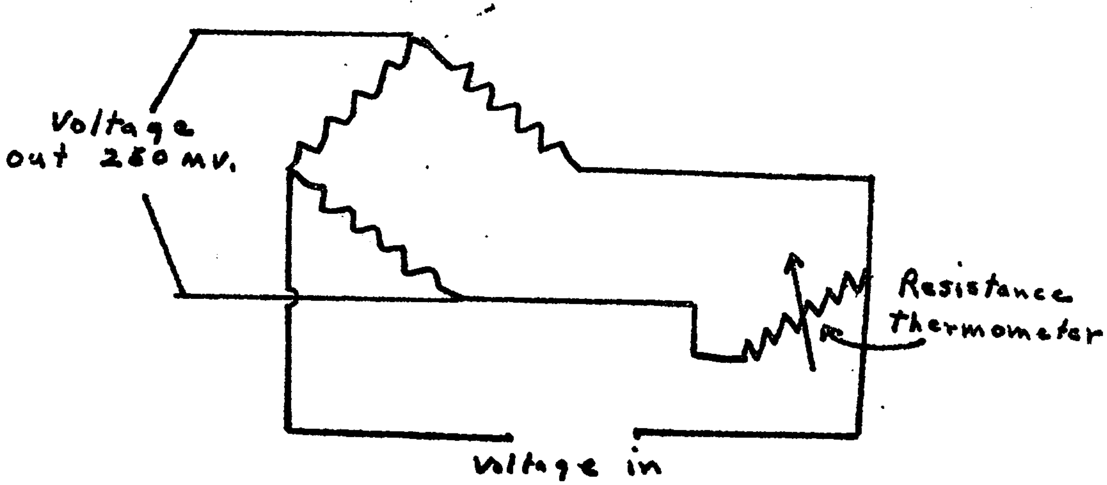
Typical Measurement Systems "Apollo"



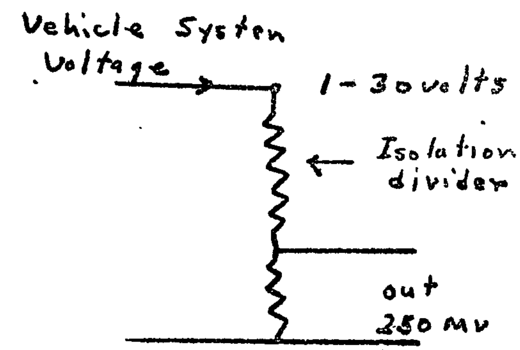
Pressure



Temperature 0 ± 1000°F



System Voltages



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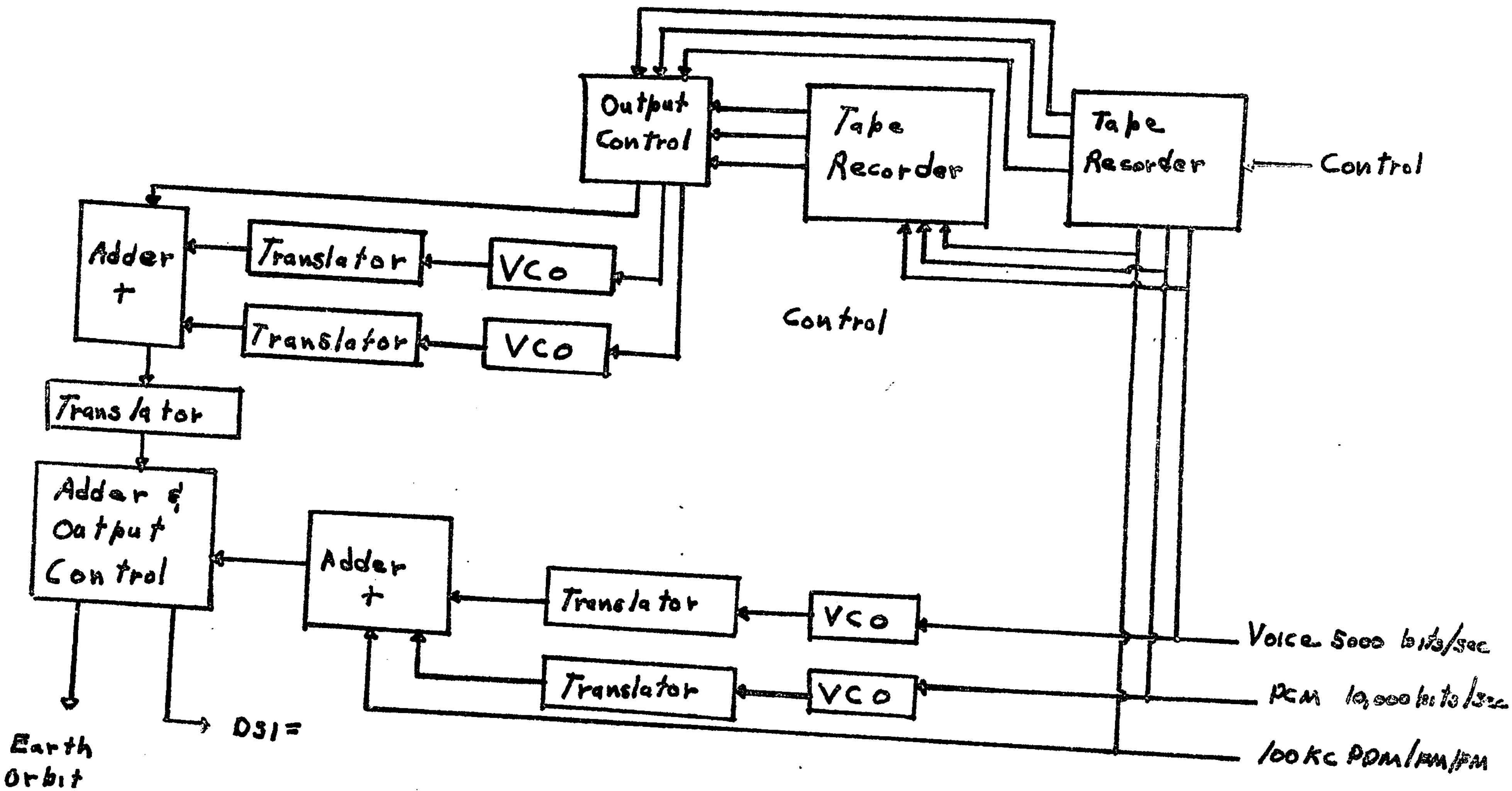
Data Translation & Recording System



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Malfunction Detection System

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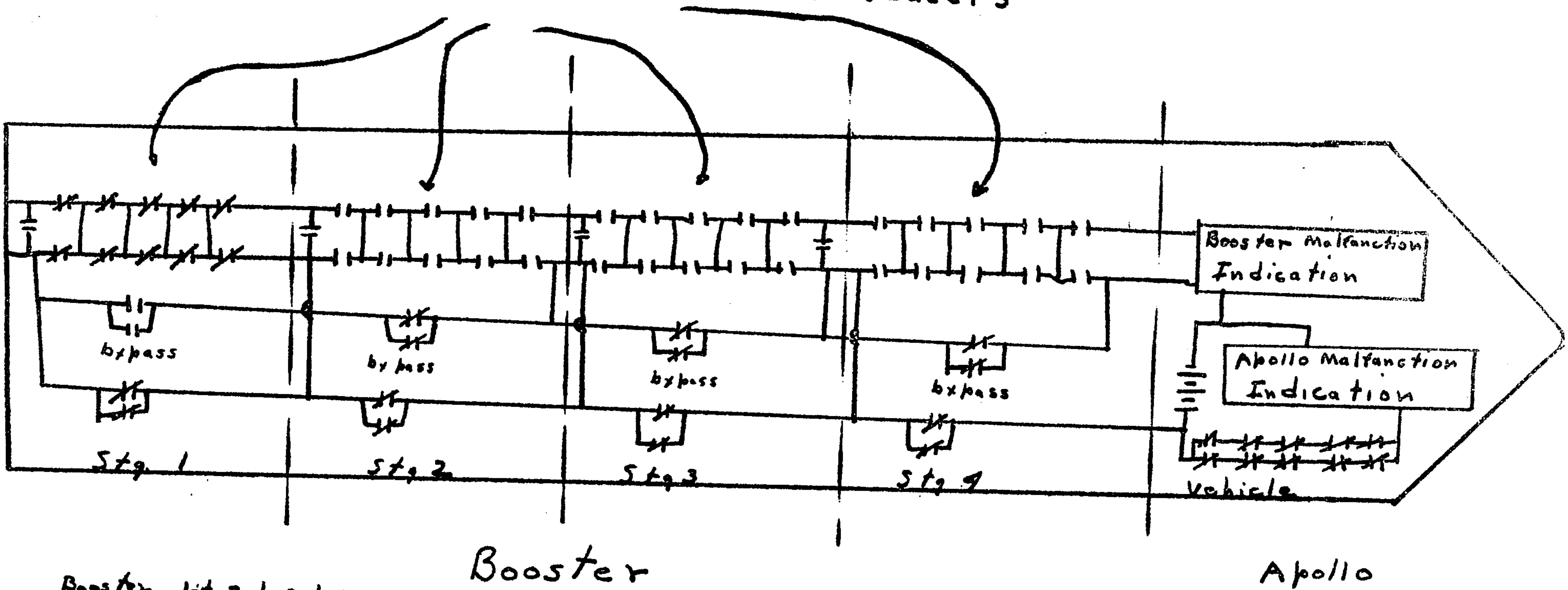
Relay contacts
⊣⊢ open
⊣/⊢ closed



Booster 1st, 2nd, 3rd & 4th stage
Functions Monitored (Typ.)

Eng. Chamber Pressures
OX Tank Pressures
Fuel Tank Pressures
Fuel Pump Discharge Press.
OX Pump Discharge Press.
Main Bus Voltage
Guidance ok
Missile Attitude

Apollo Functions
Monitored (Typ.)

Cabin Pressure
Radiation level
Total Radiation Dose
G level
Propulsion System
Guidance System
Electrical System
Ablative Heating

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